CS-59-020

CONVAIR 880

OPERATION MANUAL

CONVAIR DIVISION OF GENERAL DYNAMICS

SAN DIEGO, CALIFORNIA

Published By
CUSTOMER SERVICE DEPARTMENT
Commercial Publications

CONVAIR SAN DIEGO
GENERAL DYNAMICS CORPORATION
San Diego, California

NOTICE

TO: NORTHEAST AIRLINES, INC.

SUBJECT: CONVERSION LIST FOR AIRPLANE REGISTRATION NUMBERS

NOTICE: INSERT THIS PAGE IMMEDIATELY FOLLOWING THE TITLE PAGE IN ALL MODEL 880 MANUALS EXCEPT THE FAA APPROVED 880 FLIGHT MANUAL.

Data contained in all Model 880 Service Manuals issued to Northeast Airlines, Inc., except the FAA Approved 880 Flight Manual, reflect TWA (Trans World Airlines Inc.) airplane registration numbers. Use the following registration number conversion list to determine data applicable to the following Northeast Airlines airplanes:

NORTHEAST REGISTRATION NUMBER	TWA REGISTRATION NUMBER	CUSTOMER MODEL NUMBER	MANUFACTURING SERIAL NUMBER
N8478H	N804TW	22-1-4	00-5
N8479H	N806TW	22-1-6	00-8
N8480H	N809TW	22-1-9	00-12
N8481H	N815TW	22-1-15	00-20
N8482H	N816TW	22-1-16	00-22
N8483H	N817TW	22-1-17	00-23

Information relative to flight and operation, maintenance and repair of the airplane, airplane wiring diagrams, component overhaul, and airplane part and assembly numbers can be found in the following publications:

PUBLICATION	MANUAL NUMBER
880 Flight Manual	CS-59-019
880 Operation Manual	CS-59-020 ✓
880 Maintenance Manual	CS-59-021
880 Structural Repair Manual	CS-59-022
880 Wiring Diagram Manual	CS-59-023
880 Overhaul Manual	CS-59-024
880 Illustrated Parts Catalog	CS-59-025

CONVAIR CUSTOMER SERVICE DEPARTMENT

INTRODUCTION

Sections 1 through 4 comprise the Airplane Flight Manual for the Convair 880 Turbojet transport.

Sections 5 through 20, the Operation Manual for the Convair 880, have been prepared as a reference guide to be used by the flight crew of the Convair 880 airplane. The Operation Manual is not intended to give detailed coverage for each airplane system but does contain sufficient information to explain the function and operation of the equipment. More detailed information regarding the airplane components and systems is available from the various repair and maintenance manuals.

In all cases of variation between the first four sections of the Flight Manual and the sections comprising the Operation Manual, the Flight Manual shall govern.

CONVAIR 880

OPERATION MANUAL

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GENERAL CONTENTS

Section 5

CRUISE PERFORMANCE DATA

THIS SECTION PROVIDED
FOR CUSTOMER CONVENIENCE.
NO DATA TO BE INSERTED IN
THIS SECTION BY CONVAIR.

Section 6

GENERAL AIRPLANE DESCRIPTION

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GENERAL AIRPLANE DESCRIPTION

GENERAL DESCRIPTION

The Convair 880 turbojet transport airplane is a high-speed, medium range, low swept wing jet transport. The airplane is of all metal construction with fully cantilevered wing and tail surfaces, a pressurized and air conditioned semi-monocoque soundproofed fuselage, four wing-pylon mounted General Electric CJ805-3 turbojet engines, and a fully retractable tricycle-type landing gear. The engines are equipped with integral sound suppressors and thrust reversing mechanisms. Dimensions are illustrated in Figure 6-1 and general arrangement in Figure 6-2.

Crew Members

The operating flight crew consists of pilot, copilot and flight engineer. An observer's station is provided in the flight compartment. The airplane normally carries three cabin attendants.

Passengers

The airplane seats 84 to 110 passengers, depending on the configuration of the cabin area. The seating arrangement, lounge or club areas and the closet space varies with the standard, coach and mixed configurations according to operator requirements.

Windows

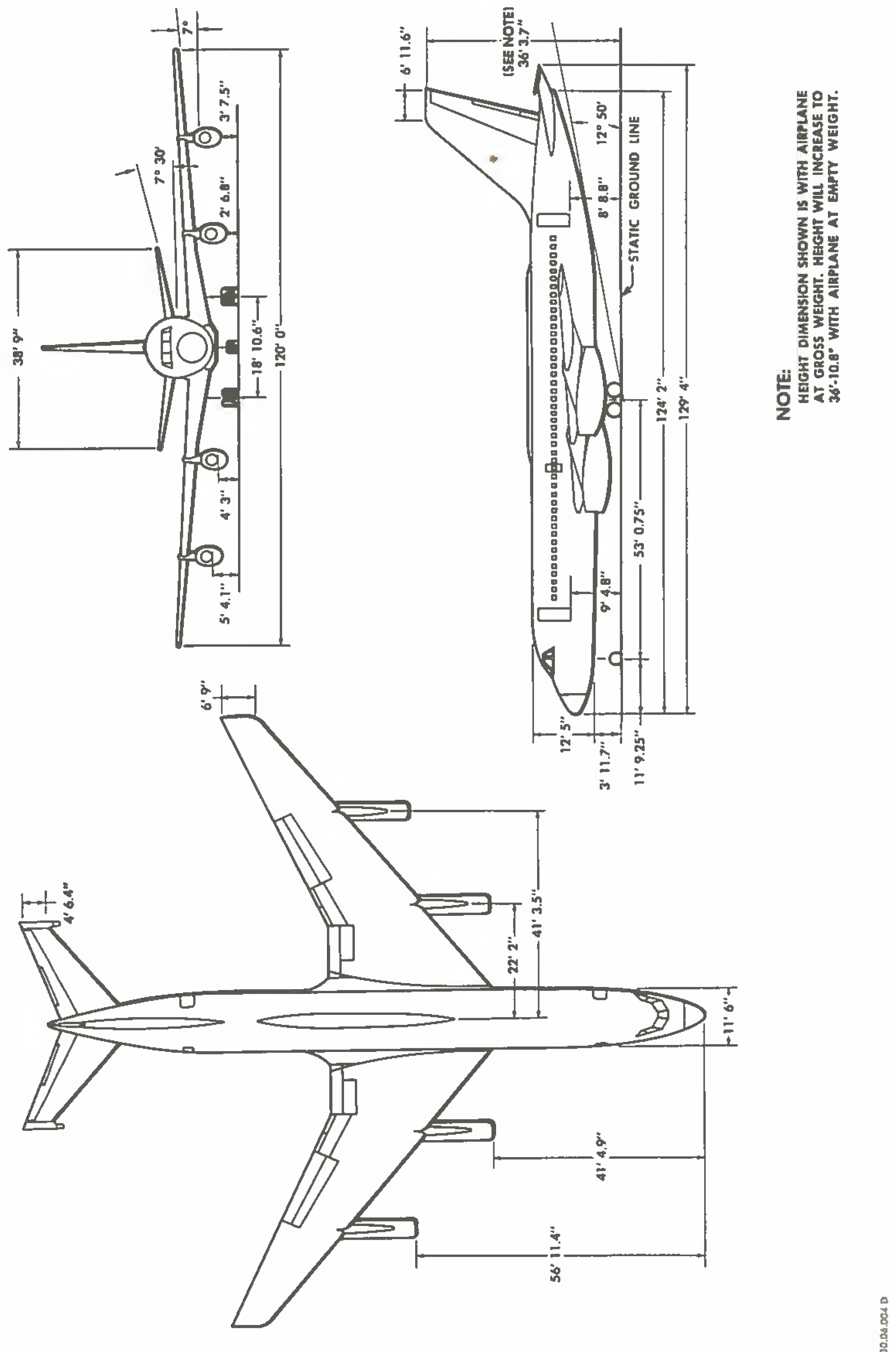
All flight compartment windows are laminated of Polyvinyl Butyral plastic and tempered plate glass. Only the three main windshields are electrically anti-iced; however, all flight compartment windows are electrically anti-fogged. Each passenger cabin window incorporates an inner and an outer window for increased blowout protection.

Doors

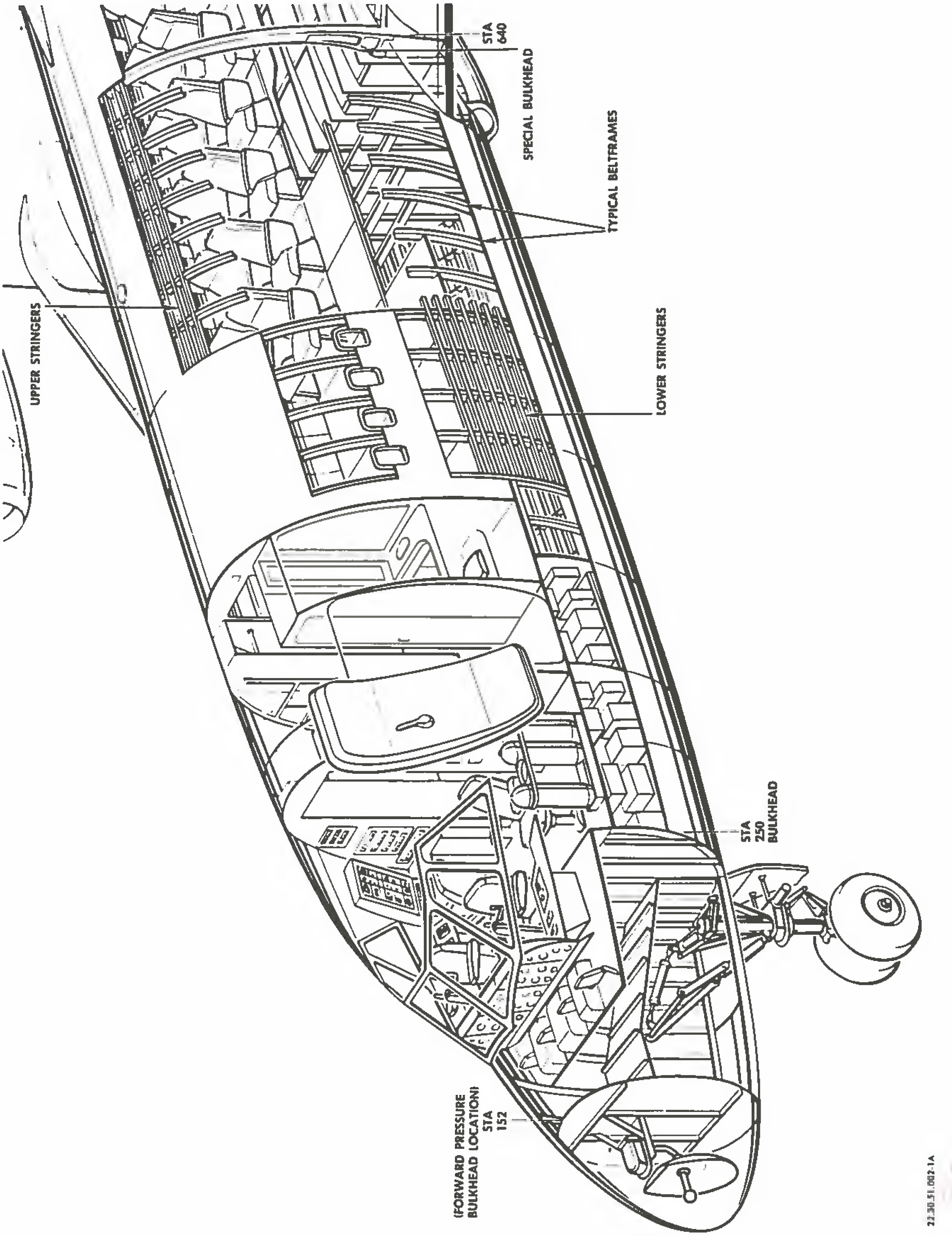
Two main entrance doors are provided on the left side of the fuselage and two service doors on the right side. The doors are wedge shaped and of the plug type construction. Rotation of either the inside or outside door handle to the open position slides the doors upward and out of the retainer tracks so that they open out and fold against the fuselage for maximum access. Other access doors are located throughout the airplane to provide ease of maintenance of the various systems and components. The access doors are shown in Figure 6-3.

Upholstery and Trim

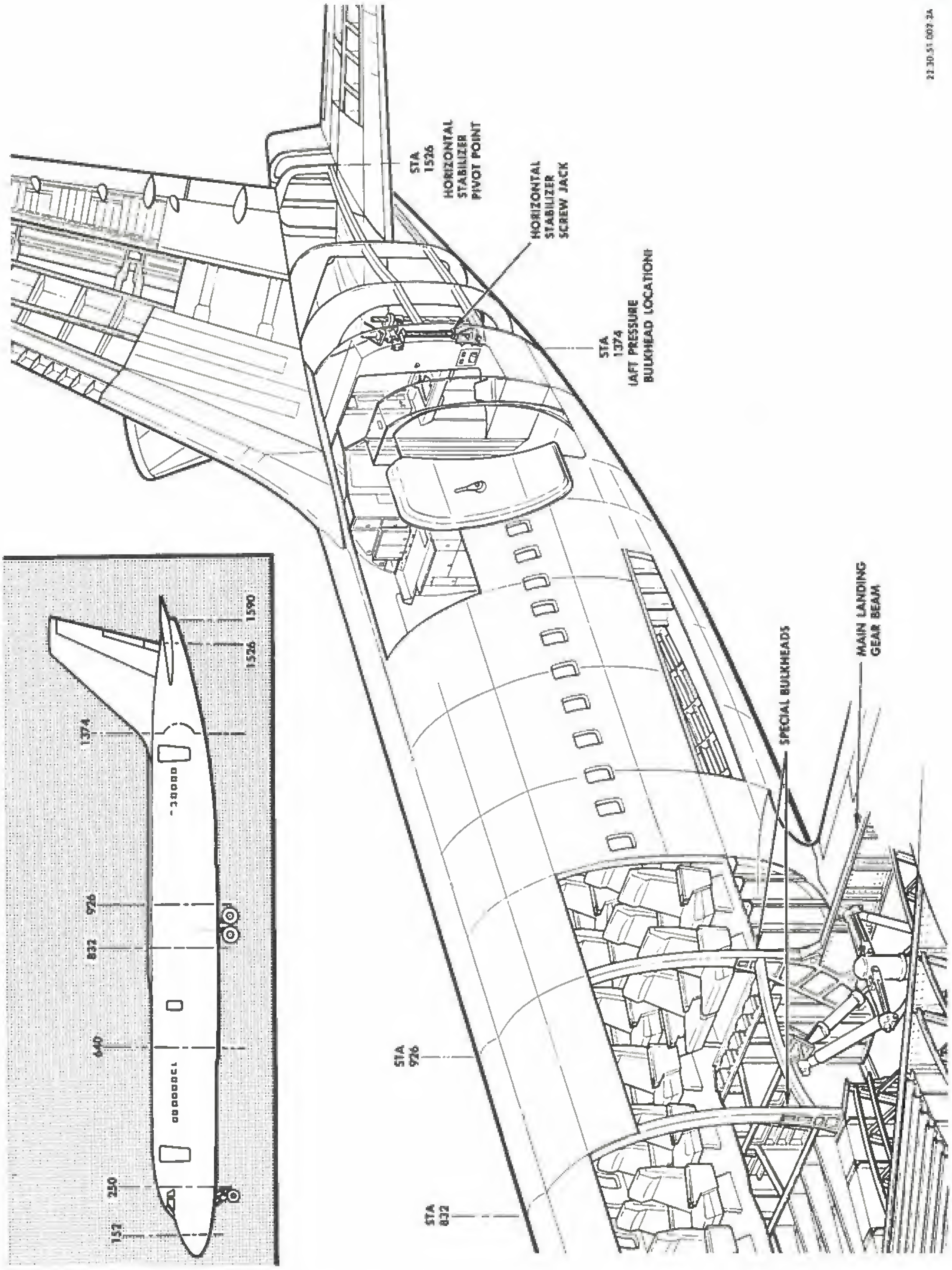
The interior is designed for beauty, comfort and ease of maintenance. The materials and fabrics used in the cabin interior will not readily stain and



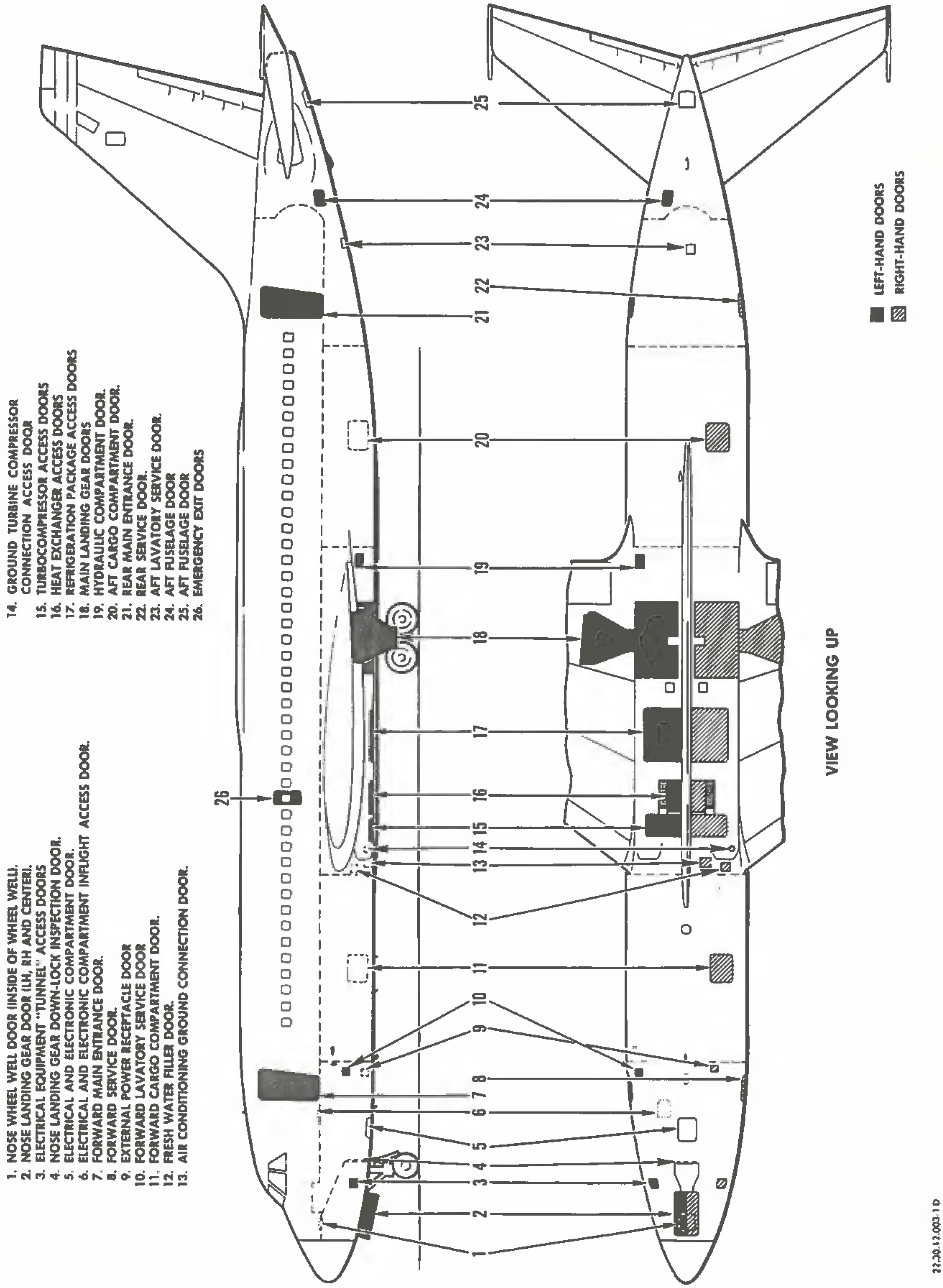
Airplane Three View
Figure 6-1



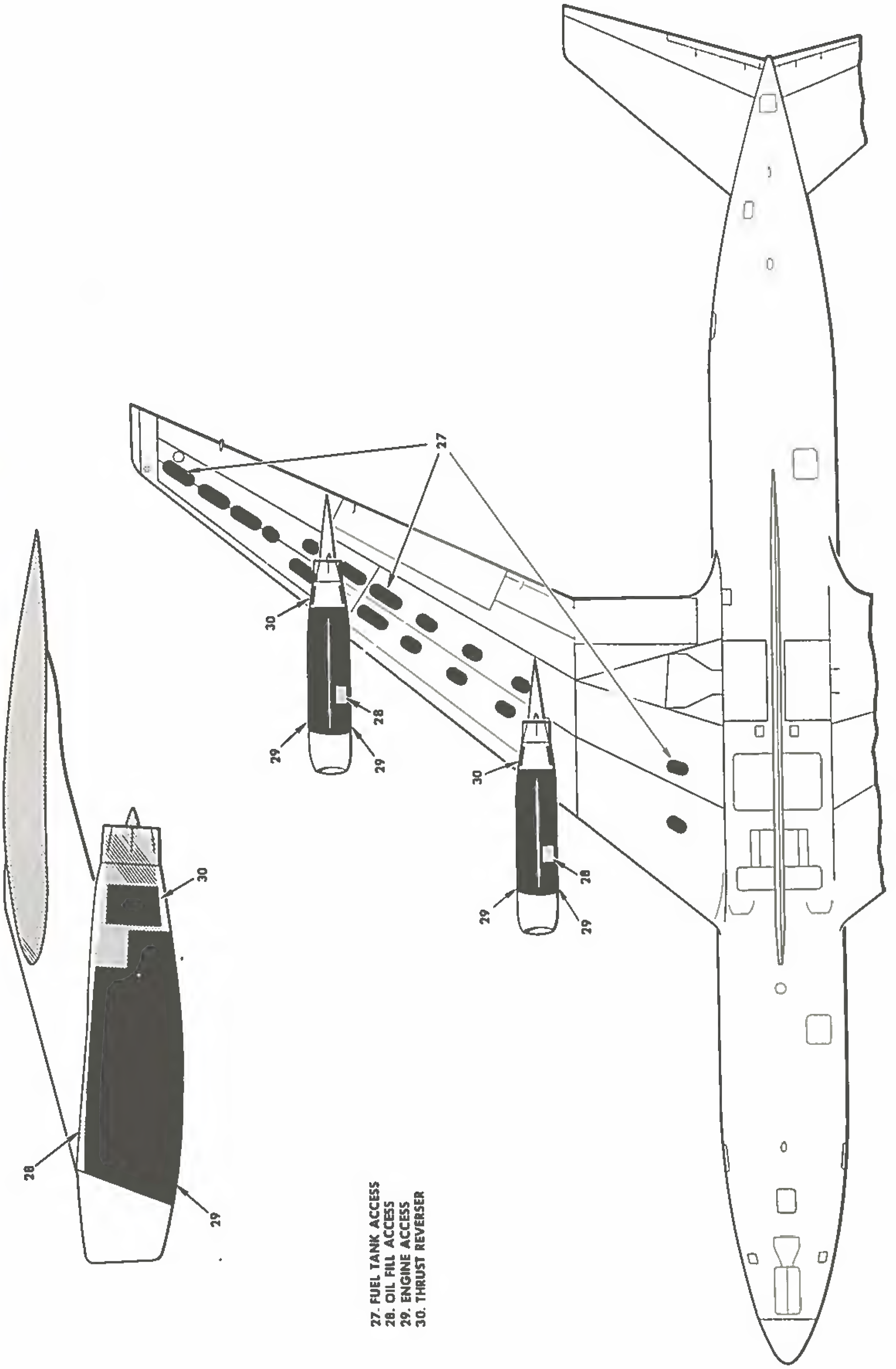
General Airplane Arrangement
Figure 6-2 (Sheet 1 of 2)



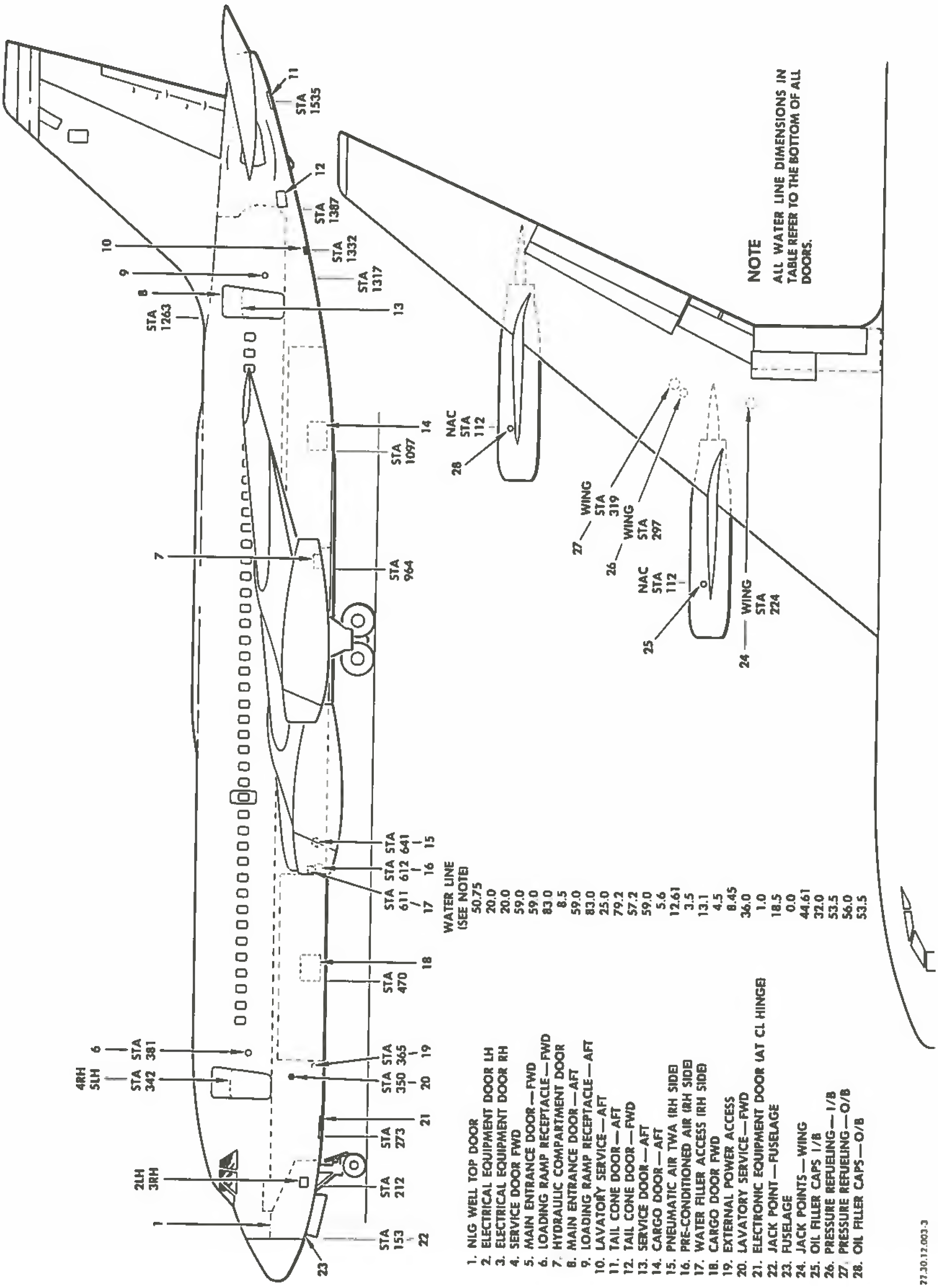
General Airplane Arrangement
Figure 6-2 (Sheet 2 of 2)



Airplane Access Doors
Figure 6-3 (Sheet 1 of 3)



Airplane Access Doors
Figure 6-3 (Sheet 2 of 3)



		WATER LINE (SEE NOTE)
1.	NLG WELL TOP DOOR	50.75
2.	ELECTRICAL EQUIPMENT DOOR LH	20.0
3.	ELECTRICAL EQUIPMENT DOOR RH	20.0
4.	SERVICE DOOR FWD	59.0
5.	MAIN ENTRANCE DOOR—FWD	59.0
6.	LOADING RAMP RECEPTACLE—FWD	83.0
7.	HYDRAULIC COMPARTMENT DOOR	8.5
8.	MAIN ENTRANCE DOOR—AFT	59.0
9.	LOADING RAMP RECEPTACLE—AFT	83.0
10.	LAVATORY SERVICE—AFT	25.0
11.	TAIL CONE DOOR—AFT	79.2
12.	TAIL CONE DOOR—FWD	57.2
13.	SERVICE DOOR—AFT	59.0
14.	CARGO DOOR—AFT	5.6
15.	PNEUMATIC AIR TWA (RH SIDE)	12.61
16.	PRE-CONDITIONED AIR (RH SIDE)	3.5
17.	WATER FILLER ACCESS (RH SIDE)	13.1
18.	CARGO DOOR FWD	4.5
19.	EXTERNAL POWER ACCESS	8.45
20.	LAVATORY SERVICE—FWD	36.0
21.	ELECTRONIC EQUIPMENT DOOR (AT CL HINGE)	1.0
22.	JACK POINT—FUSELAGE	18.5
23.	FUSELAGE	0.0
24.	JACK POINTS—WING	44.61
25.	OIL FILLER CAPS I/B	32.0
26.	PRESSURE REFUELING—I/B	53.5
27.	PRESSURE REFUELING—O/B	56.0
28.	OIL FILLER CAPS—O/B	53.5

NOTE
ALL WATER LINE DIMENSIONS IN TABLE REFER TO THE BOTTOM OF ALL DOORS.

Airplane Access Doors
Figure 6-3 (Sheet 3 of 3)

can be easily cleaned with soap and water. Fire resistant material has been used throughout the airplane.

Drainage System

A positive draining system is included to drain all areas where water or moisture might collect and also in lines carrying flammable fluids. Drained fluids can not re-enter the airplane, tailpipe, heating, anti-icing and ventilating ducts, or any other potential fire sources (see Figure 6-4).

Nacelles and Pylons

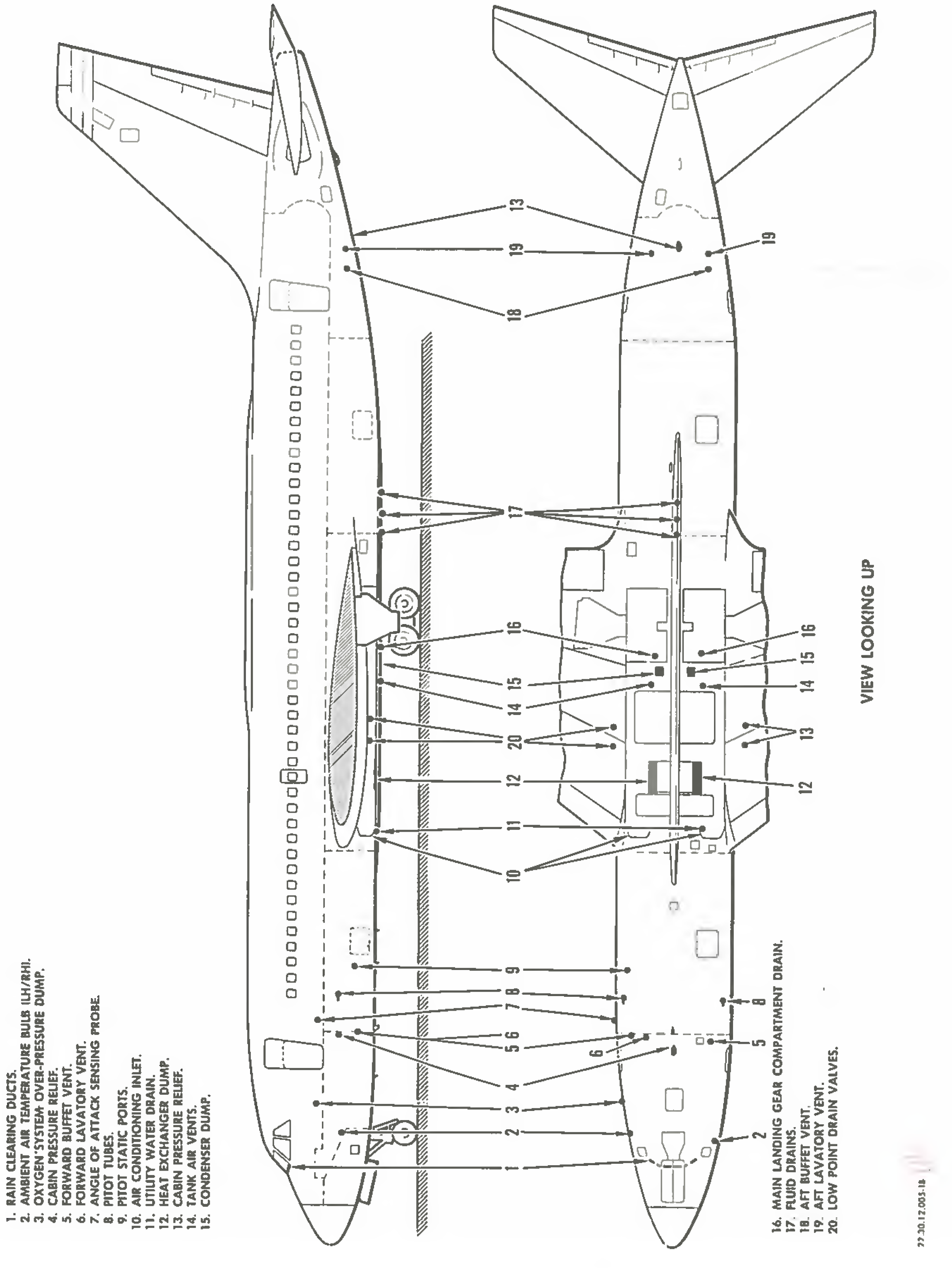
The four General Electric CJ805-3 torbojet engines are enclosed in pods attached to the lower surface of the wing by means of pylons. The pylons are of aluminum alloy construction with steel fittings at the wing and pod attaching points. The pods are constructed of aluminum alloy, steel, and titanium, depending on the fire potentials in each pod area (see Section 17, FIRE PROTECTION SYSTEM, for detailed information). Clamshell cowl doors provide full accessibility to the engines during maintenance.

Air Conditioning and Pressurization

The air conditioning and pressurization system is designed to supply all occupied compartments with an airflow of 120 pounds per minute at an airplane altitude of 41,000 feet. Circulating air, either heated or cooled, is supplied as required by the temperature control system. The air conditioning system is composed of two independent systems, either of which can furnish adequate air conditioning and pressurization for both the passenger cabin and flight compartment. Heating and cooling of the baggage compartments and the electrical and electronic equipment is also accomplished by the air conditioning system. The pressurization system can maintain a sea level cabin altitude up to an airplane altitude of 21,000 feet, and an 8000 foot cabin altitude up to an airplane altitude of 41,000 feet. The maximum normal cabin differential operating pressure is 8.3 psi. Integral warning systems are provided to indicate any system malfunctions.

Automatic Flight Control

An automatic flight control system is provided consisting of an automatic pilot, flight instrumentation, and speed stability augmentation which controls the airplane attitude, altitude and orientation. The system also provides automatic guidance under localizer, glide path, and omnirange radio control. System instruments, displays and controls are integrated for ease of control and simplicity.

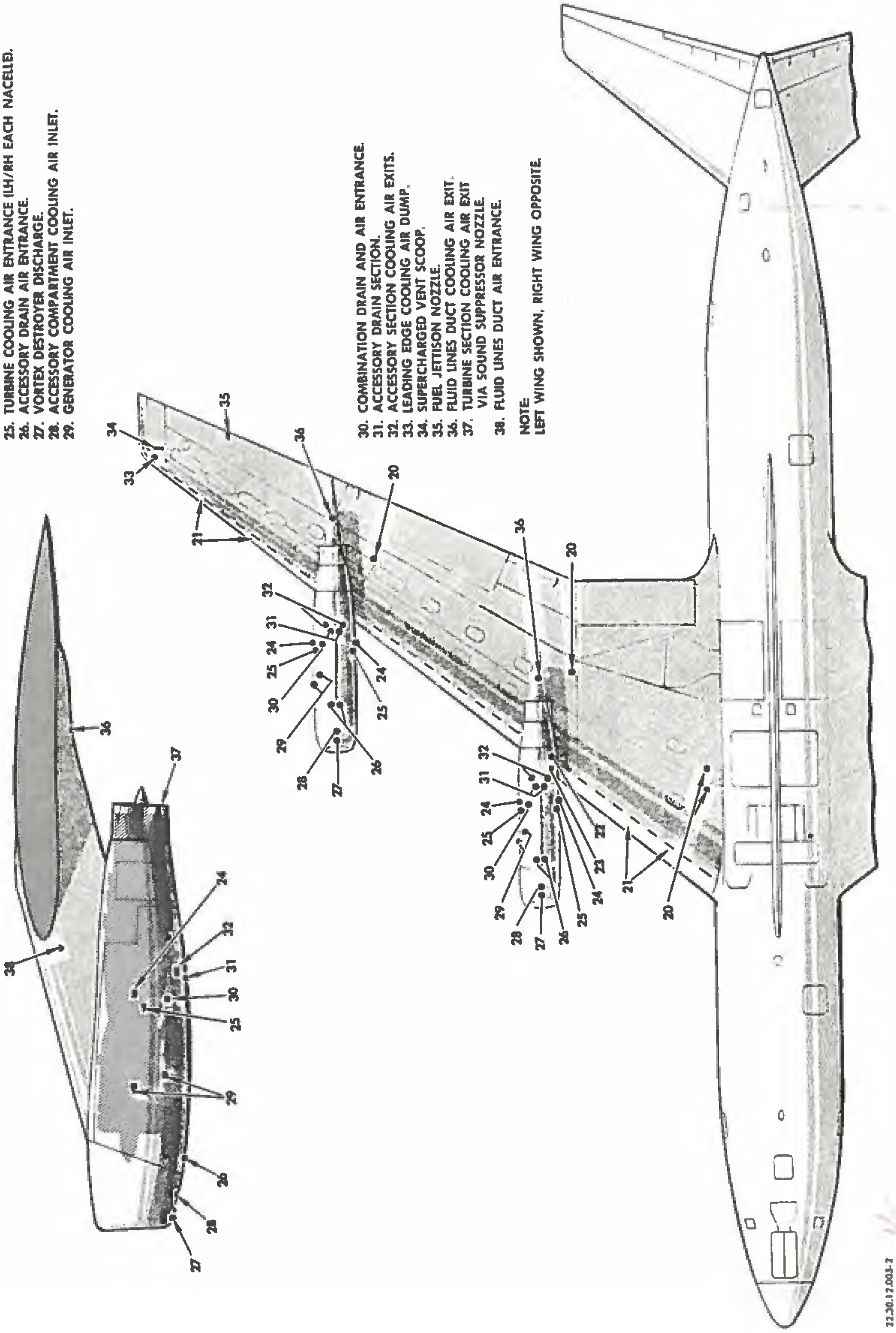


Airplane Vents and Drains
Figure 6-4 (Sheet 1 of 2)

20. LOW POINT DRAIN VALVES.
21. WING ANTI-ICE AIR DISCHARGE.
22. AIR INTAKE MOISTURE DRAIN.
23. LEADING EDGE COOLING AIR DUCT.
24. AIR LEAKAGE EXIT (LH/RH EACH NACELLE).
25. TURBINE COOLING AIR ENTRANCE (LH/RH EACH NACELLE).
26. ACCESSORY DRAIN AIR ENTRANCE.
27. VORTEX DESTROYER DISCHARGE.
28. ACCESSORY COMPARTMENT COOLING AIR INLET.
29. GENERATOR COOLING AIR INLET.

30. COMBINATION DRAIN AND AIR ENTRANCE.
31. ACCESSORY DRAIN SECTION.
32. ACCESSORY SECTION COOLING AIR EXITS.
33. LEADING EDGE COOLING AIR DUMP.
34. SUPERCHARGED VENT SCOOP.
35. FUEL JETTISON NOZZLE.
36. FLUID LINES DUCT COOLING AIR EXIT.
37. TURBINE SECTION COOLING AIR EXIT VIA SOUND SUPPRESSOR NOZZLE.
38. FLUID LINES DUCT AIR ENTRANCE.

NOTE:
LEFT WING SHOWN, RIGHT WING OPPOSITE.



Airplane Vents and Drains
Figure 6-4 (Sheet 2 of 2)

Flight Instruments

A Kollsman KS-86 integrated flight instrument system (KIFIS) incorporates integral scale error correctors in the instruments and compensates for errors due to static pressure variations. Individual systems are provided for the pilot and copilot.

Electrical Power Supply

The electrical power supply system provides adequate power for the operation of all electrical and electronic circuits with a built-in power reserve. Alternating current is supplied by four engine-mounted ac generators driven by constant speed drives. Four transformer-rectifier units supply normal 28-volt dc power to all dc buses when the ac system is energized. A single aircraft type battery supplies emergency dc power to the emergency bus when required. The battery remains on a floating charge when not in use.

Flight Controls

The flight control system consists of a primary and secondary group. The primary group includes the rudder, ailerons, elevators and spoilers; the secondary group is composed of the trim tabs for the primary controls, horizontal stabilizer trim, flaps and speed brakes. The primary controls are aerodynamically operated by flight tabs positioned by the control column, wheel and rudder pedals. The flaps are divided into two sections on each side of the fuselage due to the sweptback wing configuration. The spoilers serve a dual function, working in conjunction with the ailerons and also as speed brakes. The inboard and outboard spoilers can be "split" by the pilot to provide either a pitch-up or pitch-down reaction, as required for emergency longitudinal trim. The horizontal stabilizer is built as a single unit and is completely adjustable for longitudinal trim. All the primary controls have integral hydraulic gust dampeners to prevent damage by winds while on the ground. These dampeners do not interfere with normal control movement.

Fuel System

The fuel system is designed to supply and control the flow of fuel to the four engines without interruption under all environmental conditions. The tanks in each wing are divided into four fuel-tight compartments. Two compartments serve as replenishing tanks to the two main tanks. Total fuel capacity is approximately 10,770 U.S. Gallons (approximately 70,000 pounds). Each wing fuel system operates independently of the other, but are connected through various indicating, refueling, dumping, venting and crossfeed systems. The engine fuel control system regulates the amount of fuel in relation to the desired rpm, compressor inlet air pressure and temperature, acceleration and deceleration rate, and jet nozzle area. Fuel from any main tank can be delivered to any or all operating engines, however, normal operation is tank-to-engine. The normal refueling procedure utilizes underwing refueling adapters. Automatic selection of any quantity up to a full tank, exclusive of the minimum allowable expansion space, is available. Overwing gravity fill can be used if desired. Dripsticks are provided.

Hydraulic Power Supply

The main hydraulic power supply system is divided into two independent power supply subsystems with a minimum of interconnection. Each is a 3000 psi, closed center, continuously operating system. Each system consists of two variable displacement engine-driven pumps, a reservoir, accumulators and various other components. The two systems are interconnected at the reservoirs to permit single point servicing, and through a pressure line from the auxiliary hydraulic pump to permit ground operations and maintenance. Ground service connections for both systems are located in the hydraulic compartment.

Adverse Weather Protection

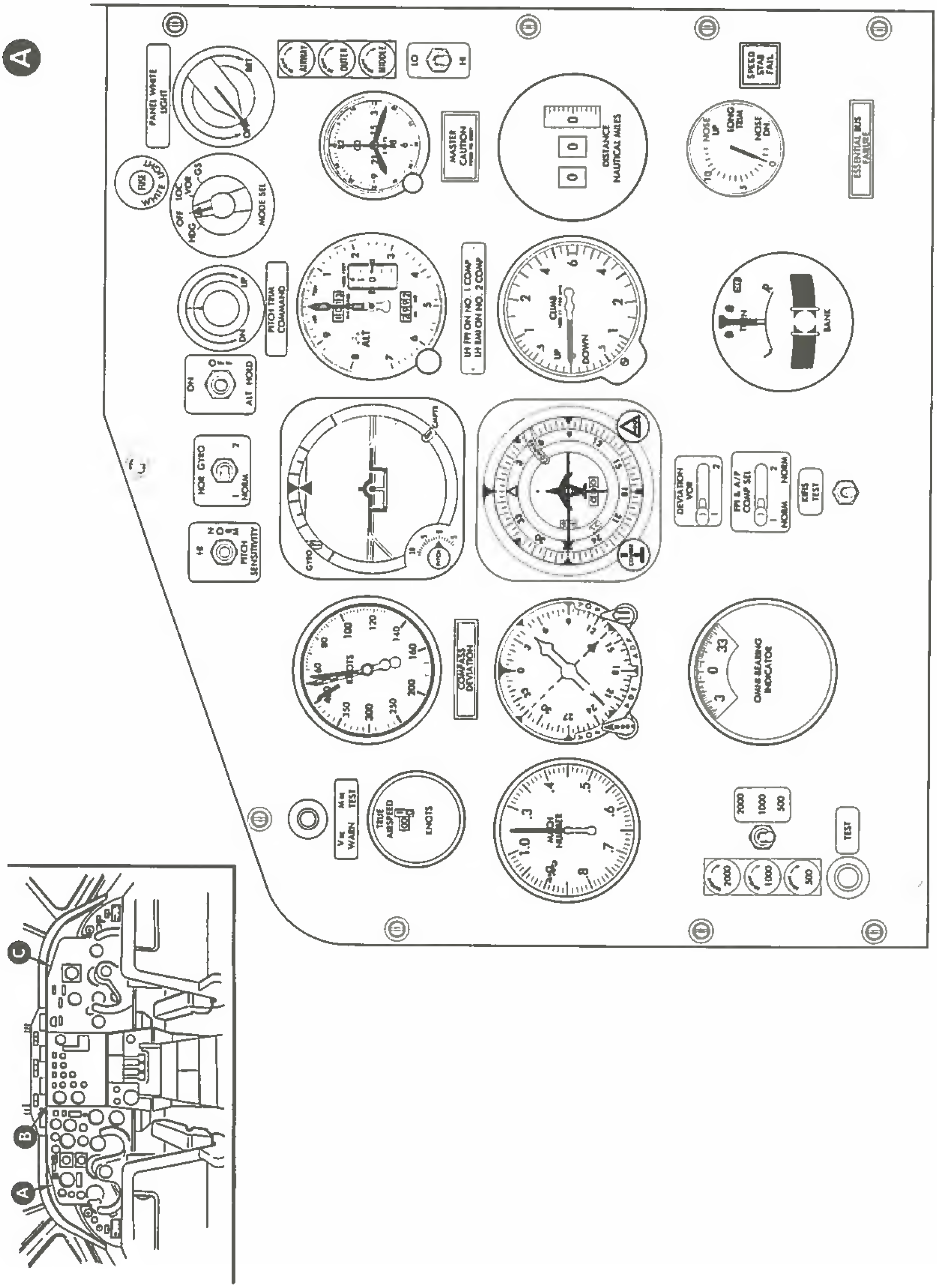
The anti-icing and de-icing systems utilize bleed air and electrical current for their operation. Bleed air flow provides heat for wing and engine anti-icing and pressure for windshield rain clearing. The horizontal and vertical stabilizer leading edges are de-iced and the windshields anti-iced and anti-fogged by electrical current. The windshields are constructed of laminated glass with conductive layers for electrical current. Operation of the adverse weather systems is automatic with provisions for manual override.

Flight Crew Instruments

Instruments have been arranged on a need basis for each of the crew members. The pilot and copilot are provided with duplicate flight instruments and various other instruments to suit their particular requirements. Engine instruments are grouped on a center panel and are readily visible to either pilot (see Figures 6-5 and 6-6). Each pilot has access to the overhead switch panel and individual console panels which contain various instruments and controls (see Figures 6-7, 6-8 and 6-9). The control panel for the flight engineer is divided into sub-panels, each of which includes all the instruments and controls for a particular airframe system (see Figure 6-10). Instruments and controls on all panels are so arranged that they can be removed or replaced without disturbing or disconnecting adjacent units.

Landing Gear

The tricycle-type landing gear is fully retractable and consists of a steerable, dual-wheel nose gear and two main gears with tandem-mounted dual wheels. The main gears retract inboard into the fuselage at the wing junction point and the nose gear retracts forward into the nose wheel well. Each gear has an integral self-adjusting brake and anti-skid system. Other features incorporated in the landing gear system include shimmy dampening, attitude positioners, brake equalizers and tubeless tires.



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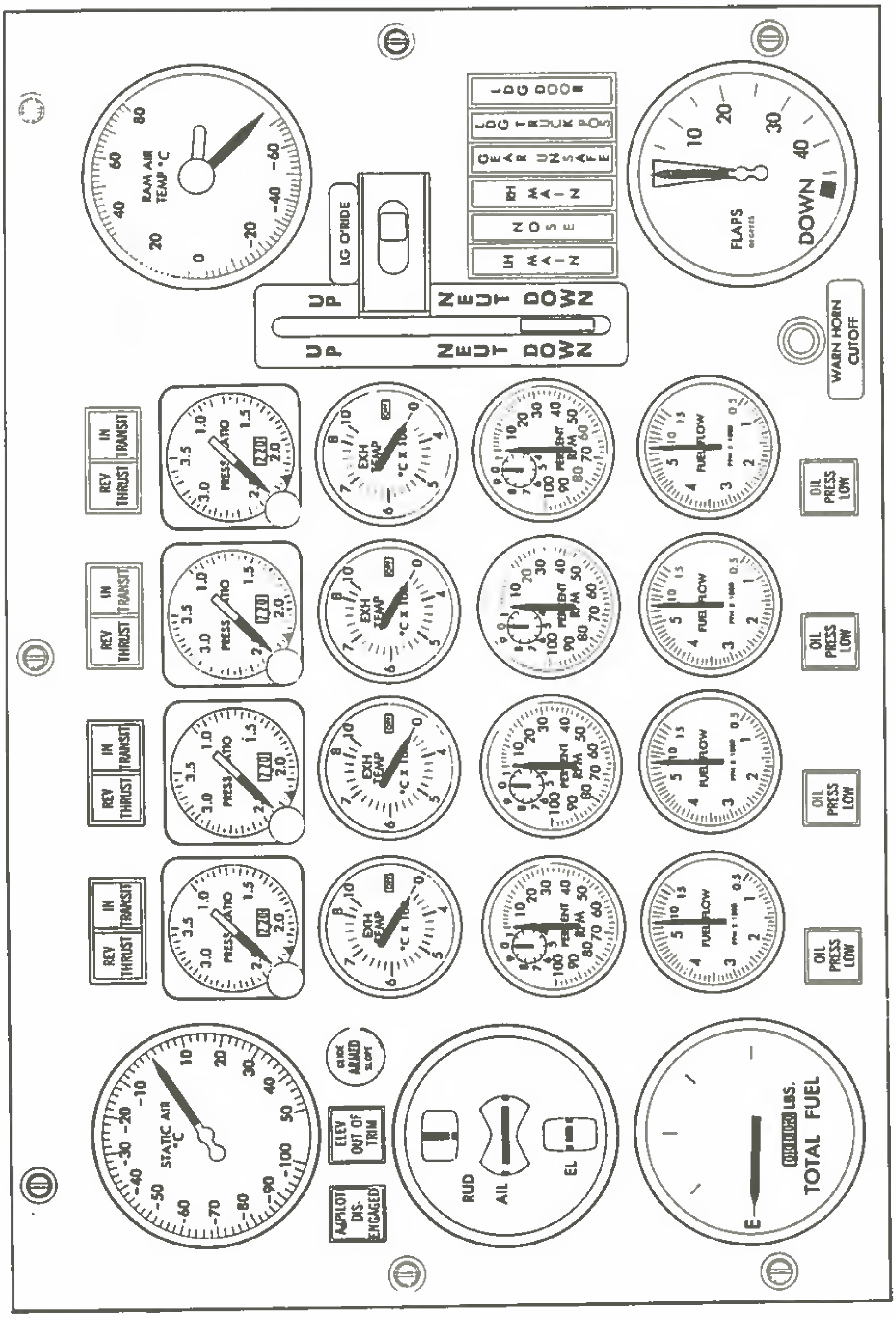
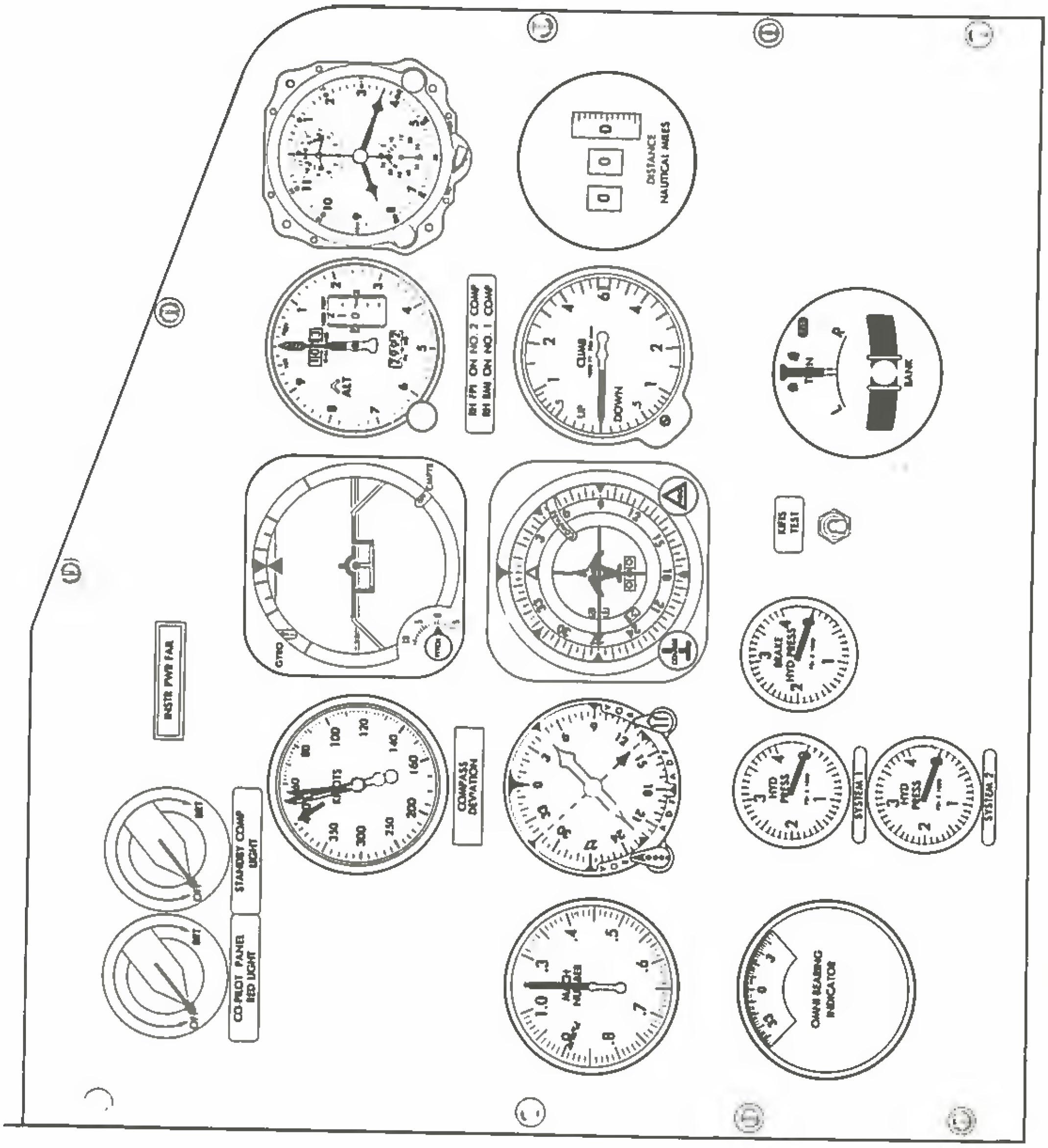
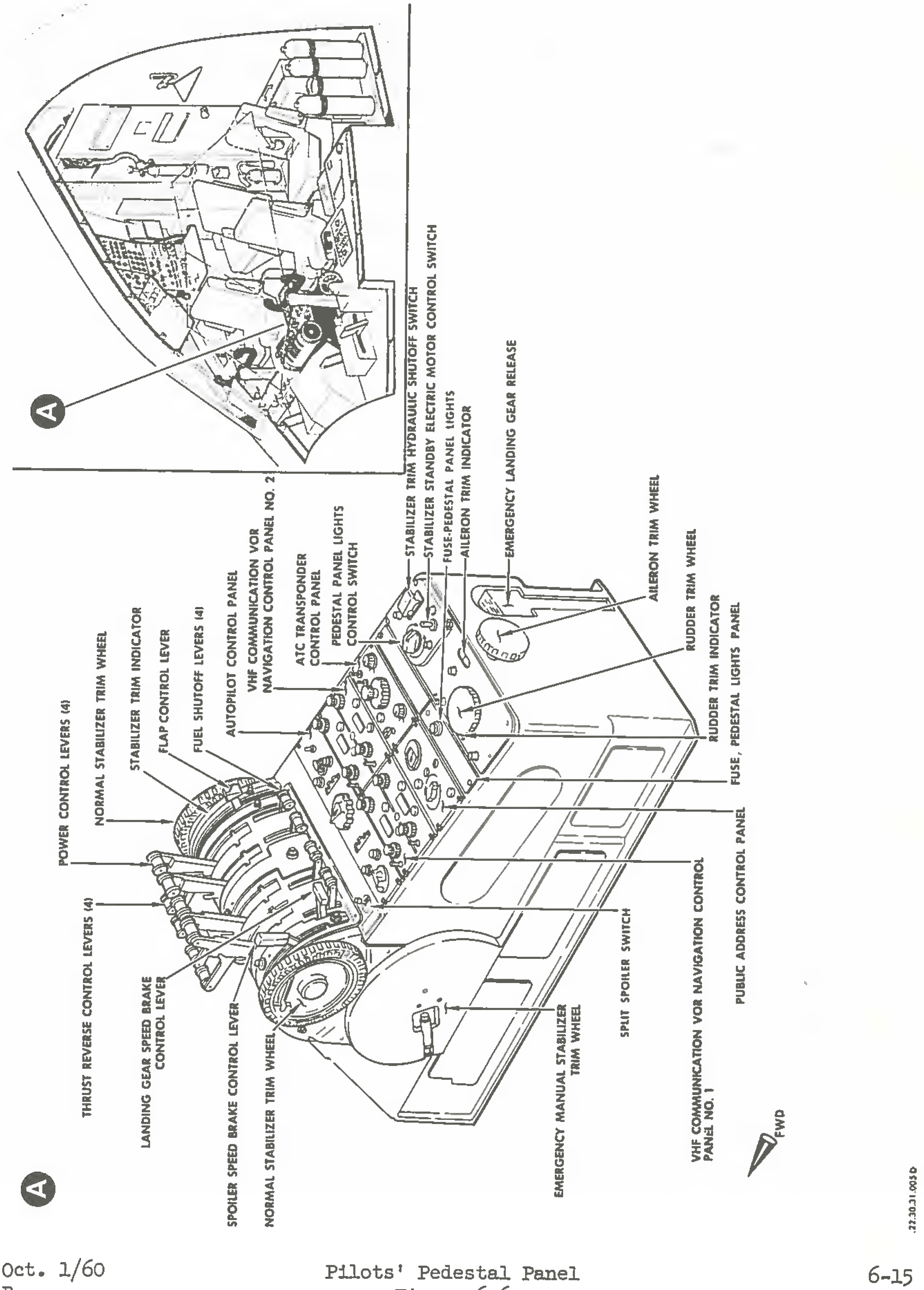
Pilots' Instrument Panels
Figure 6-5 (Sheet 1 of 3)



Pilots' Instrument Panels
Figure 6-5 (Sheet 2 of 3)

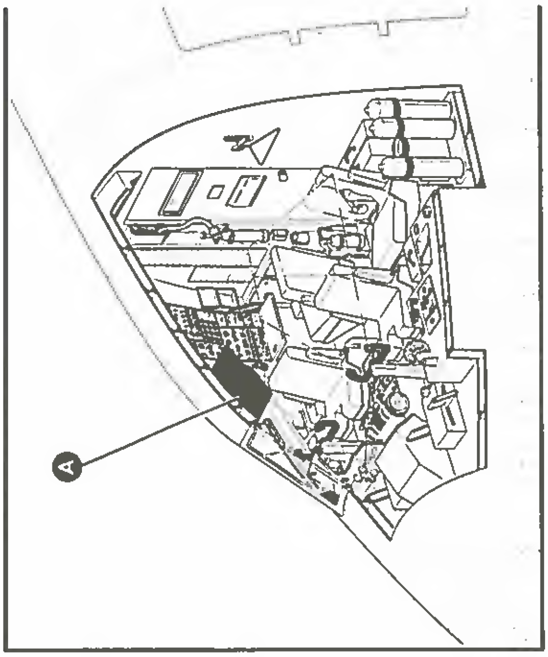


Pilots' Instrument Panels
Figure 6-5 (Sheet 3 of 3)

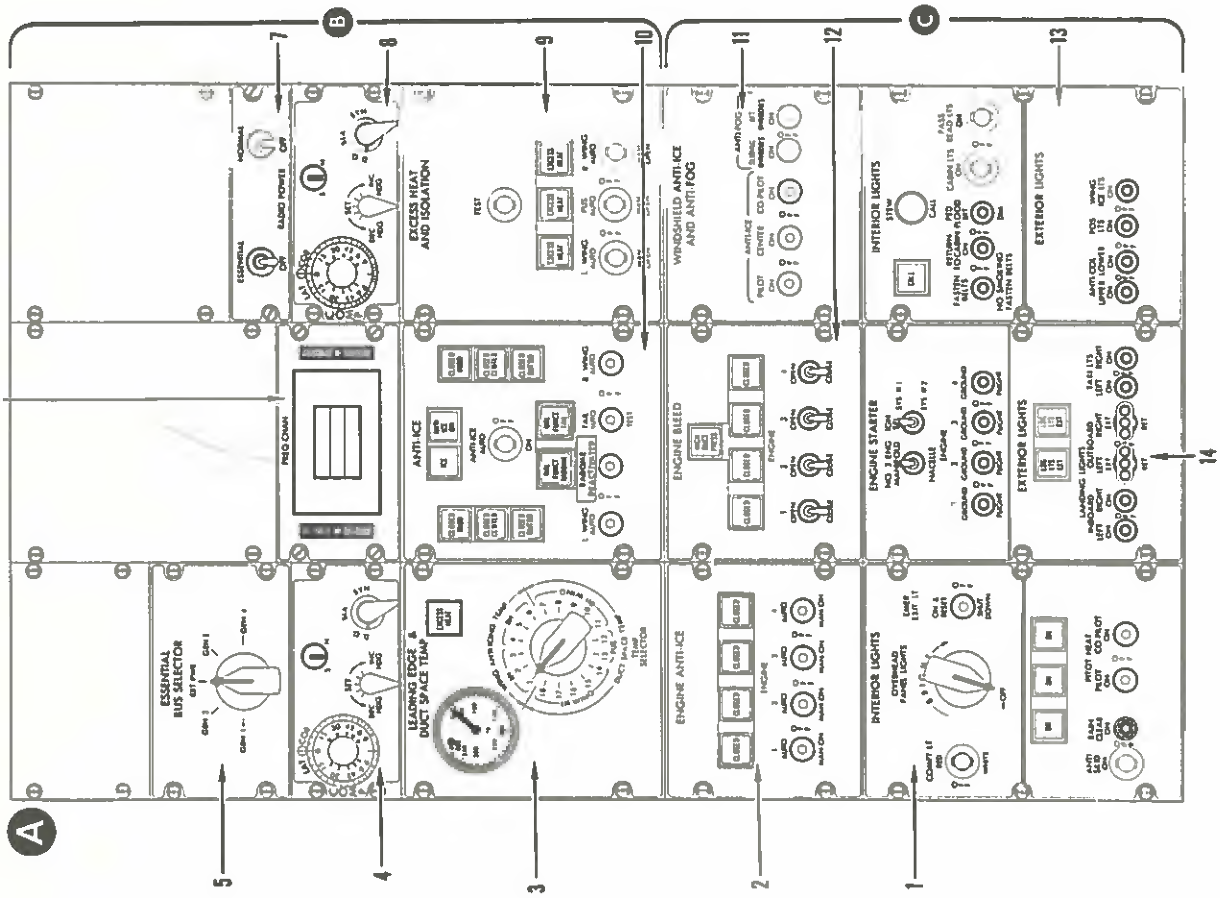


Pilots' Pedestal Panel
Figure 6-6

1. INTERIOR LIGHTS AND MISCELLANEOUS CONTROL PANEL
2. ENGINE ANTI-ICE CONTROL PANEL
3. LEADING EDGE AND DUCT SPACE TEMPERATURE PANEL
4. POLAR PATH COMPASS CONTROL PANEL
5. PILOTS ESSENTIAL BUS SELECTOR PANEL
6. RADIO FREQUENCY CHART PANEL
7. RADIO POWER PANEL
8. POLAR PATH COMPASS CONTROL



9. EXCESS HEAT AND ISOLATION CONTROL PANEL
10. WING AND TAIL ANTI-ICE CONTROL PANEL
11. WINDSHIELD ANTI-ICE AND ANTI-FOG CONTROL PANEL
12. ENGINE BLEED CONTROL PANEL
13. INTERIOR AND EXTERIOR LIGHTS CONTROL PANEL
14. ENGINE STARTER AND EXTERIOR LIGHTS CONTROL PANEL



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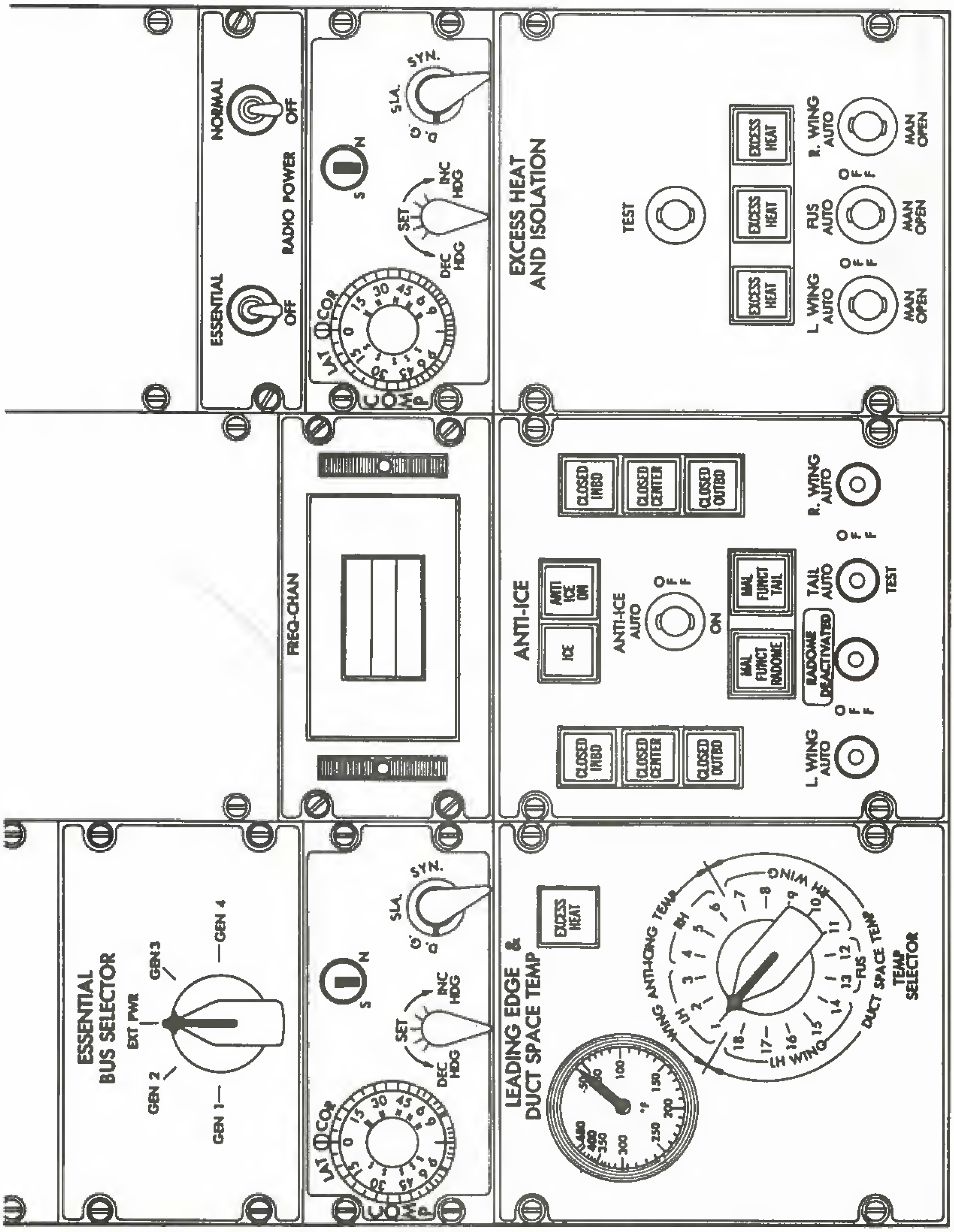
Pilots' Overhead Switch Panel
Figure 6-7 (Sheet 1 of 3)



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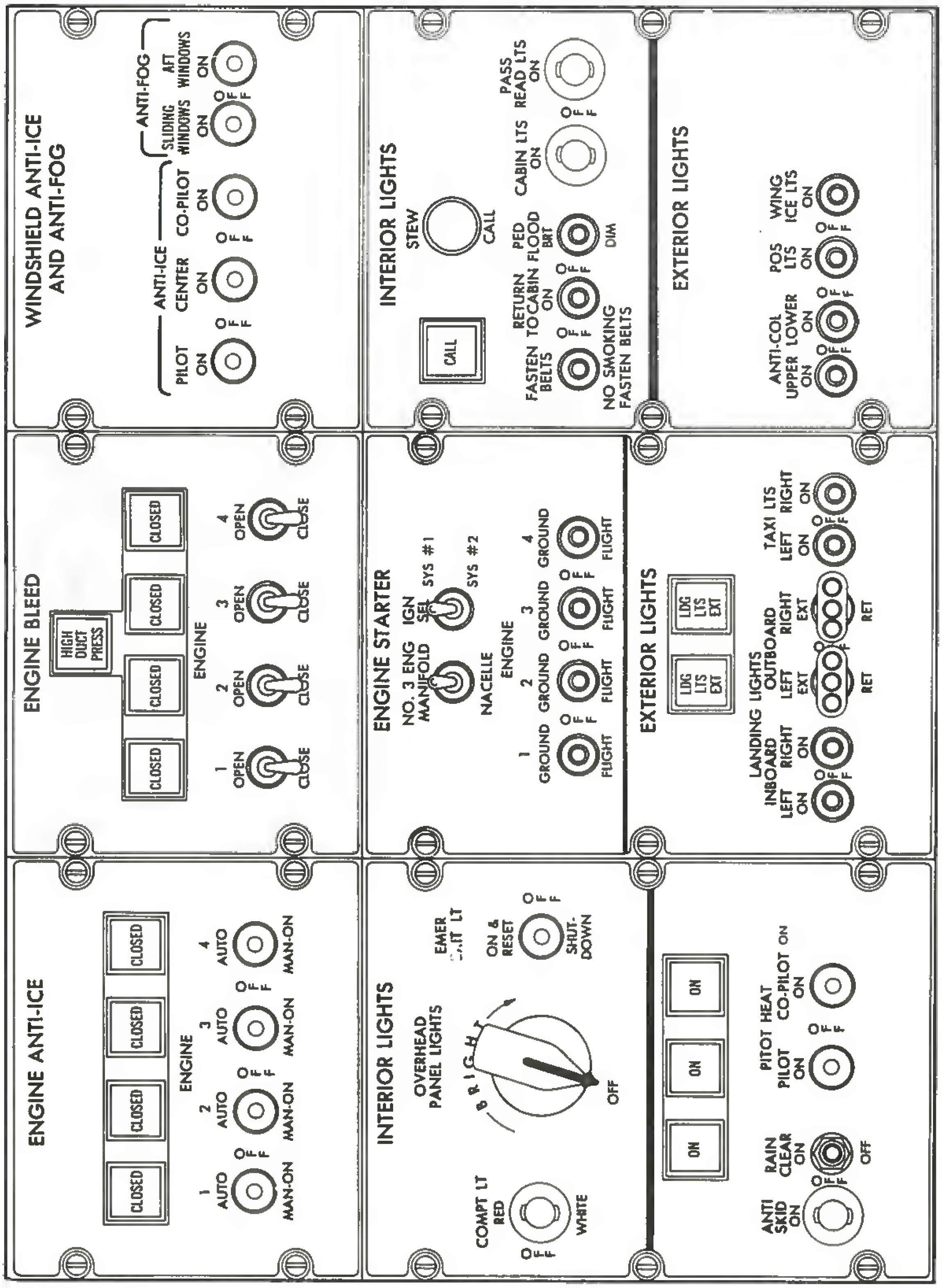
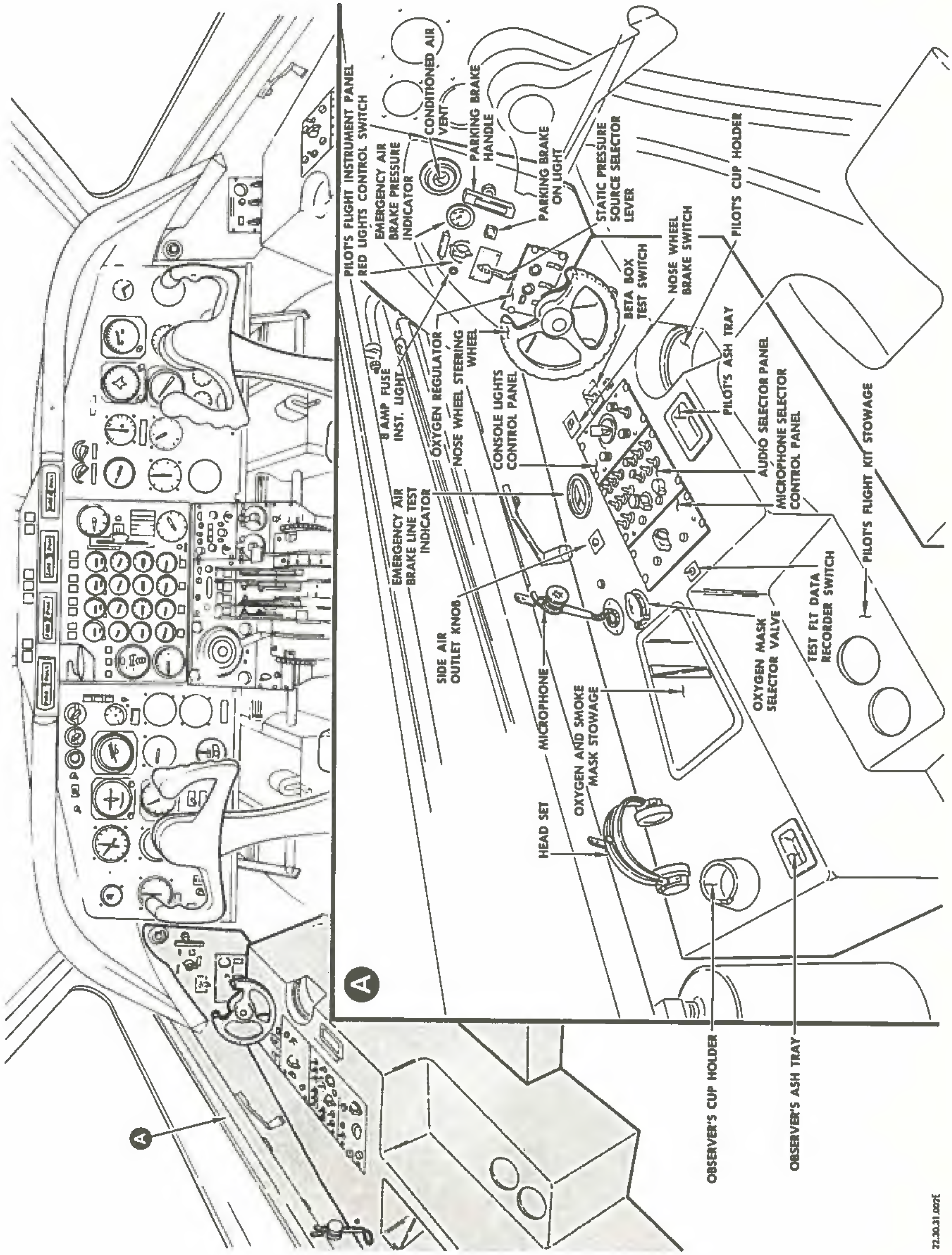
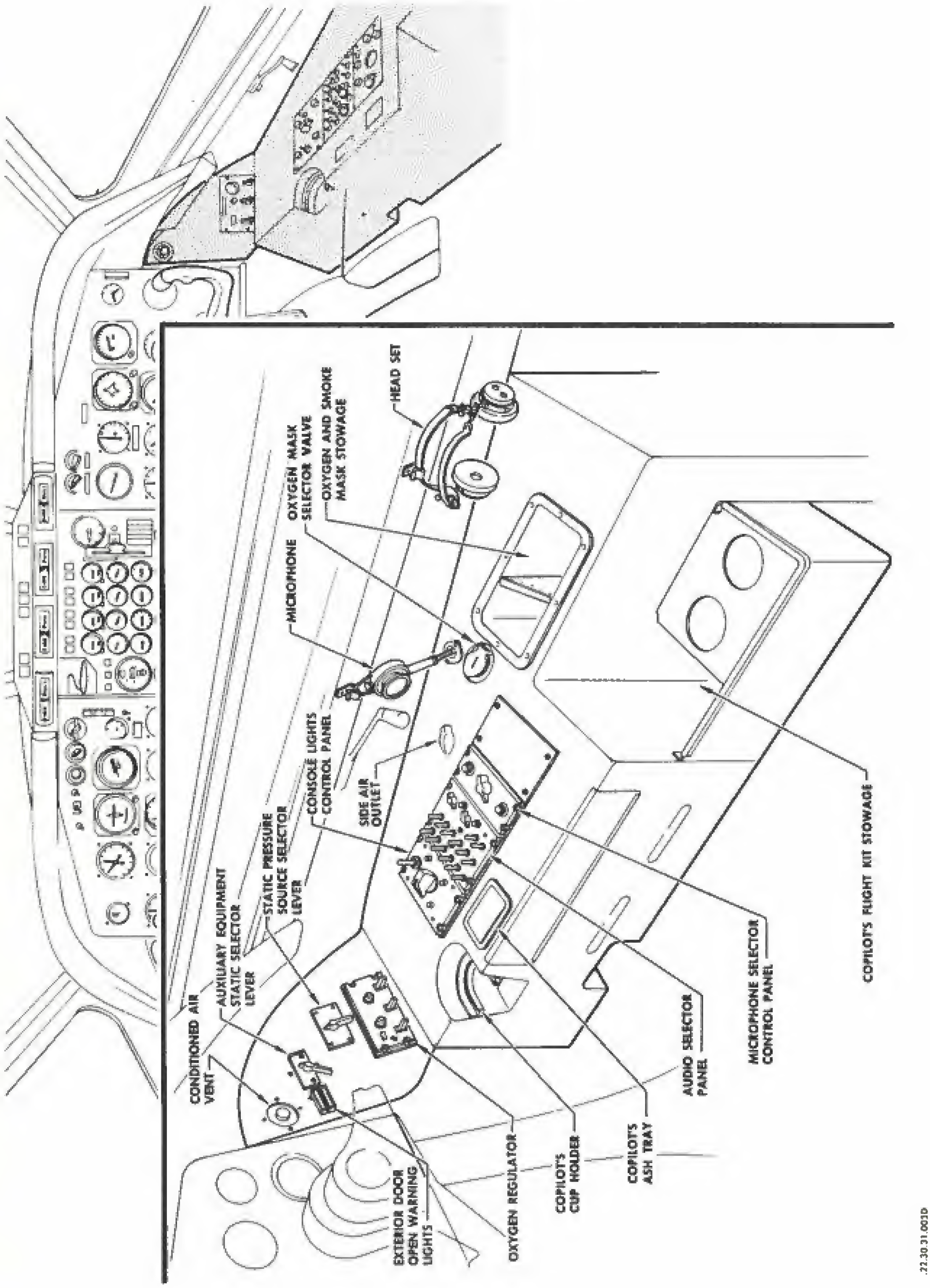
Pilots' Overhead Switch Panel
Figure 6-7 (Sheet 2 of 3)



Pilots' Overhead Switch Panel
Figure 6-7 (Sheet 3 of 3)

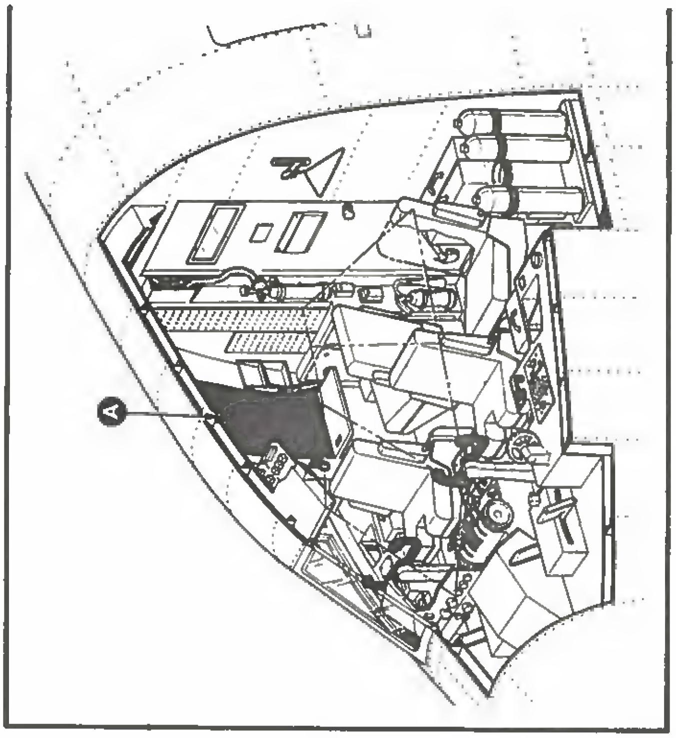


Pilot's Auxiliary Instrument and Console Panels
Figure 6-8

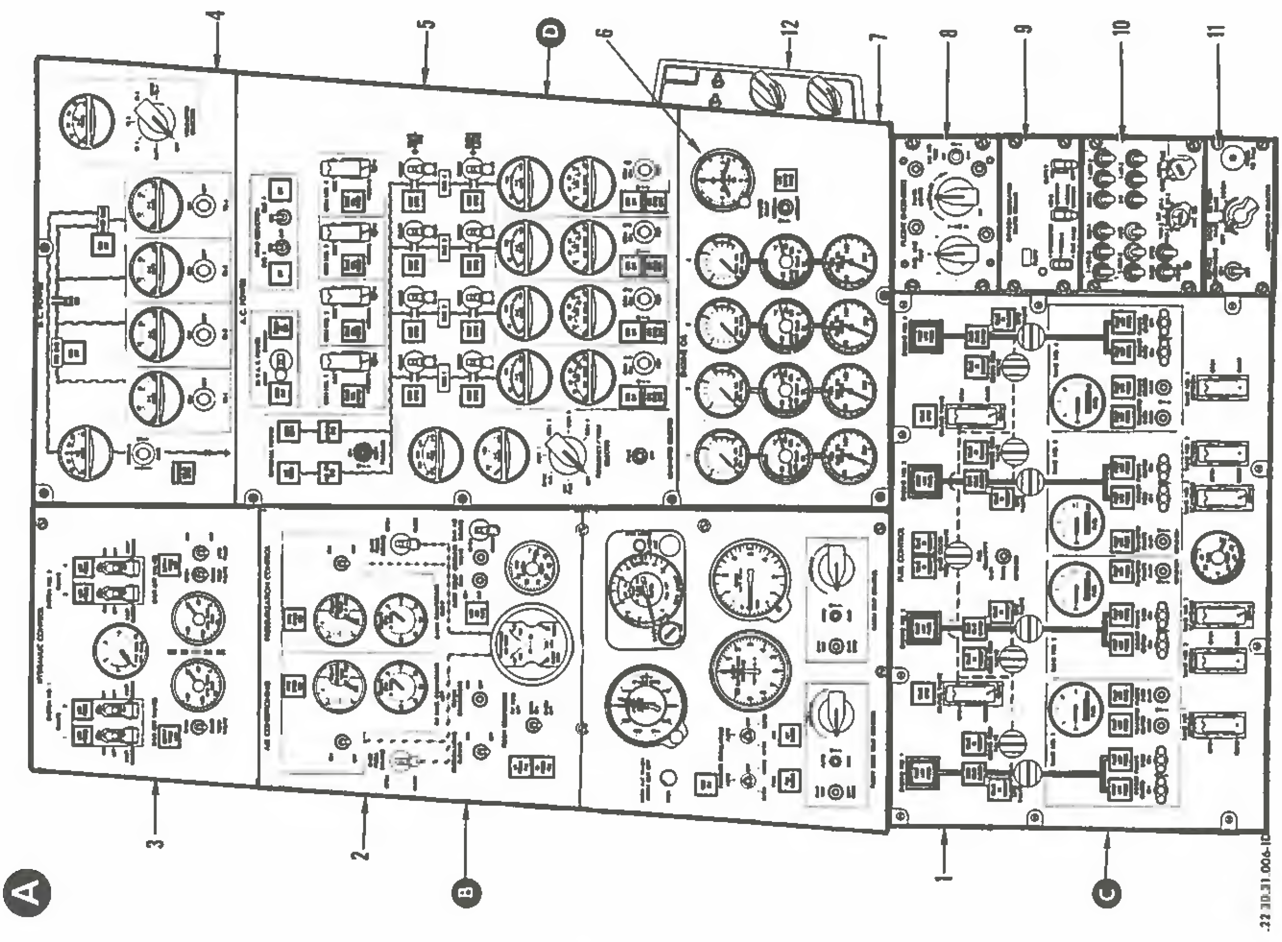


Copilot's Auxiliary Instrument and Console Panels
Figure 6-9

1. FUEL CONTROL PANEL
2. AIR CONDITIONING AND PRESSURIZATION CONTROL PANEL
3. HYDRAULIC CONTROL PANEL
4. DC POWER CONTROL PANEL



5. AC POWER CONTROL PANEL
6. CLOCK
7. ENGINE OIL INSTRUMENT PANEL
8. PANEL LIGHTS CONTROL PANEL
9. OXYGEN REGULATOR
10. AUDIO SELECTOR PANEL
11. MICROPHONE SELECTOR PANEL
12. FLOODLIGHT CONTROL PANEL



Flight Engineer's Instrument Panel
Figure 6-10 (Sheet 1 of 4)

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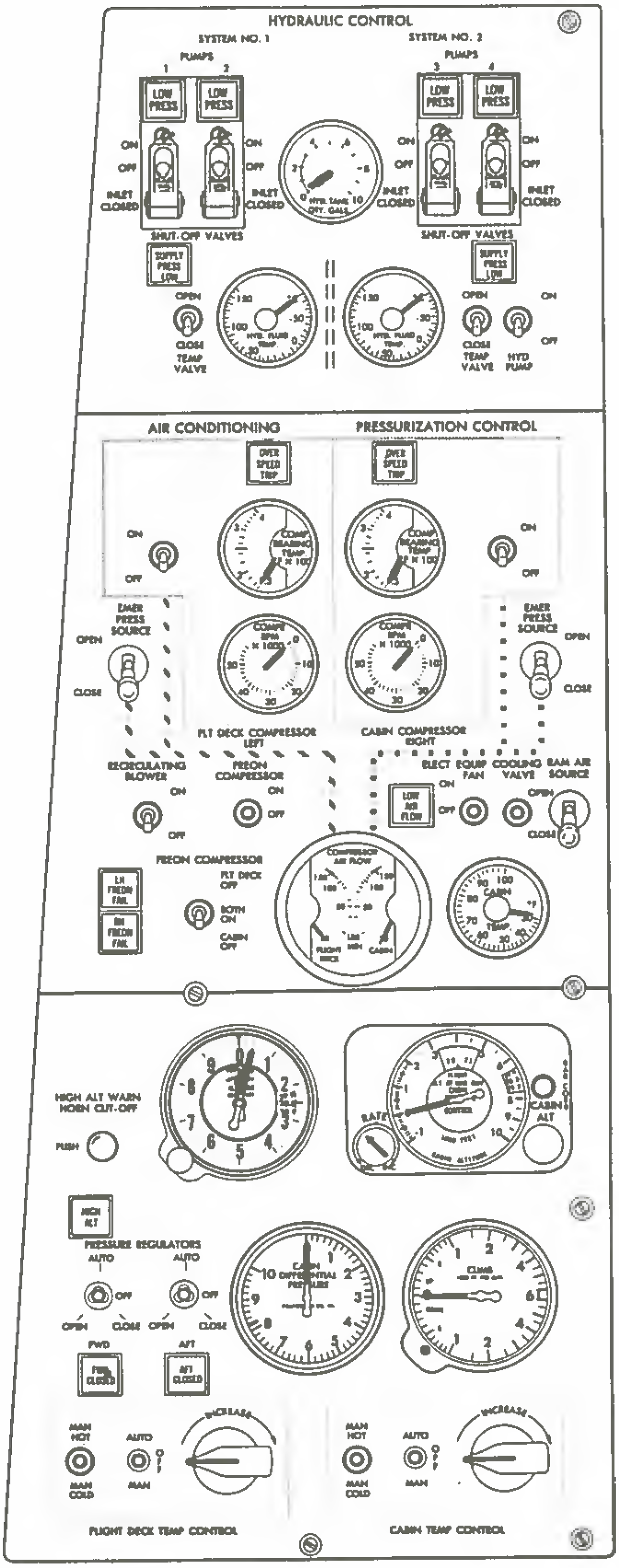

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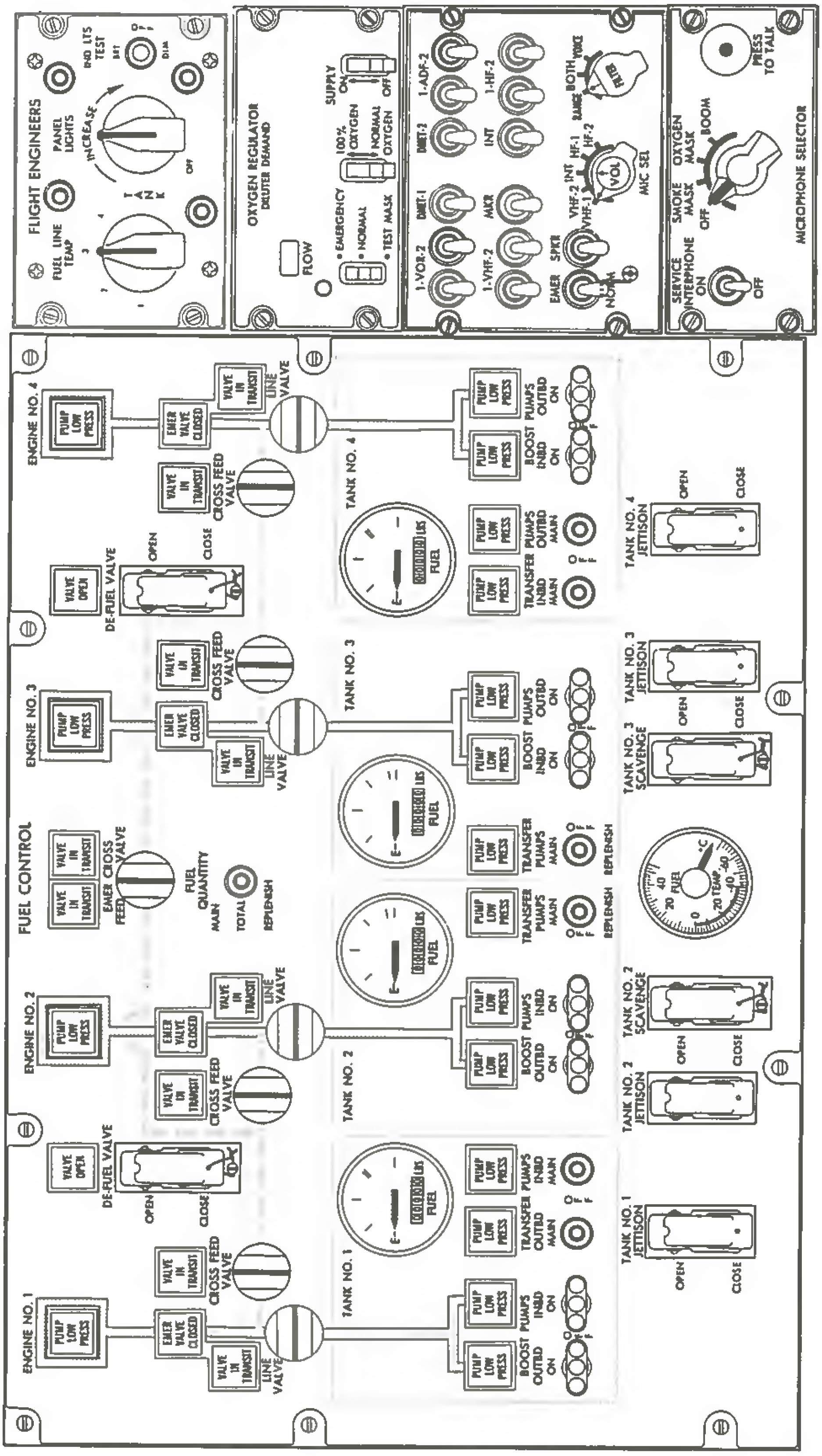
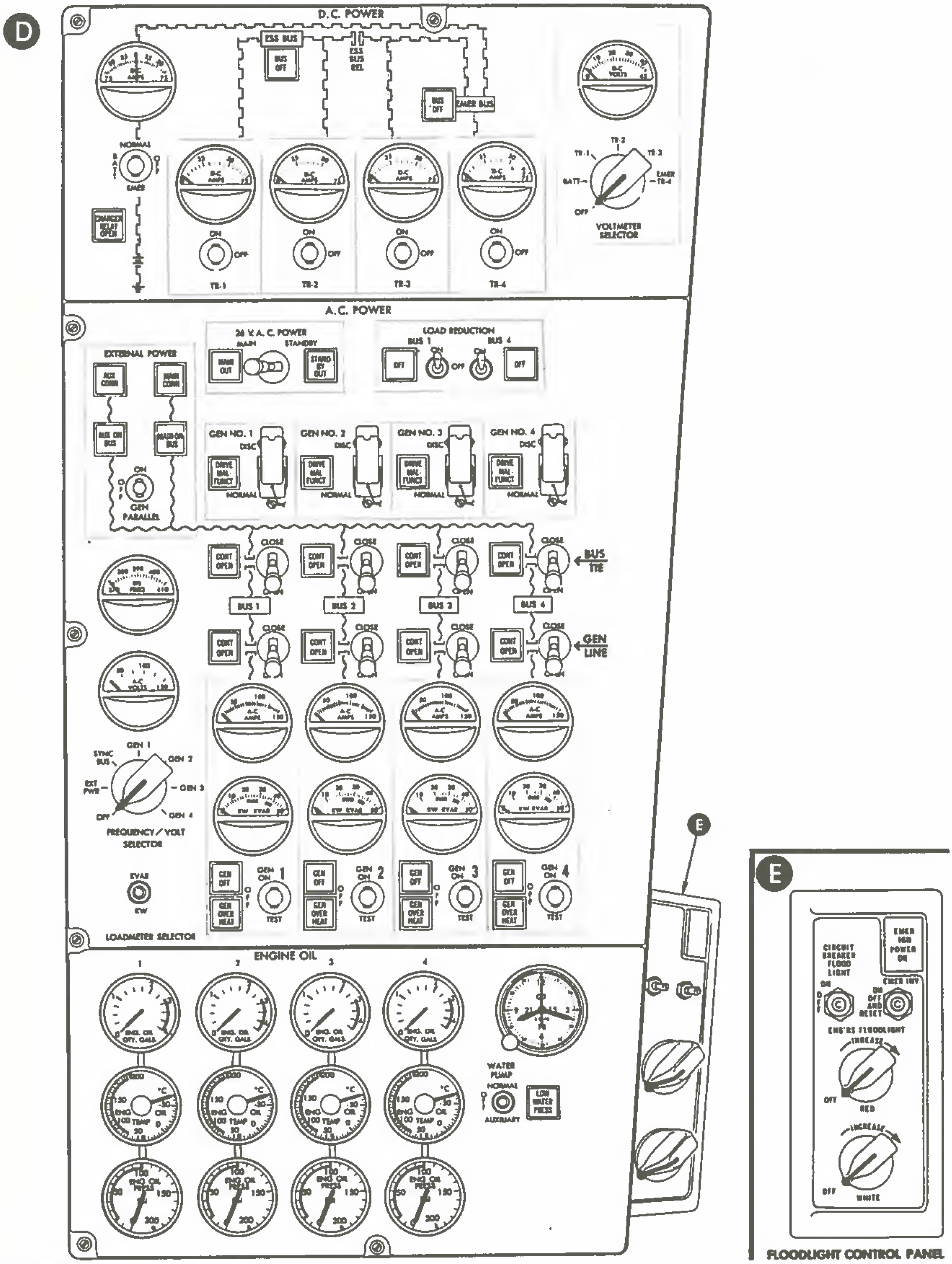
Flight Engineer's Instrument Panel
Figure 6-10 (Sheet 2 of 4)



Flight Engineer's Instrument Panel
Figure 6-10 (Sheet 3 of 4)



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Flight Engineer's Instrument Panel
Figure 6-10 (Sheet 4 of 4)

Oxygen System

The oxygen system is a high pressure, gaseous type system. Oxygen is supplied from light weight steel cylinders, each having a capacity of 107 cubic feet at a pressure of 1800 psi. The airplane also carries portable oxygen cylinders to augment the main system. The flight crew is supplied by a diluter-demand type subsystem, while the passengers, cabin attendants and observer are supplied by a continuous-flow subsystem. Release of oxygen masks and oxygen flow to the passenger compartments is automatic with manual override provisions.

Pneumatics

High pressure air is utilized in the landing gear system to provide an emergency brake system. Two other air flasks located in the hydraulic compartment serve as an air supply for emergency opening of the main gear doors.

Water and Waste Systems

A fresh water system is provided for use in the buffet and lavatory areas. Water for the system is supplied from a water tank located below the cabin floor. Sump tanks for water waste are located in the lavatory compartments, and draining and filling accommodations are easily accessible for servicing.

DIMENSIONS AND AREAS

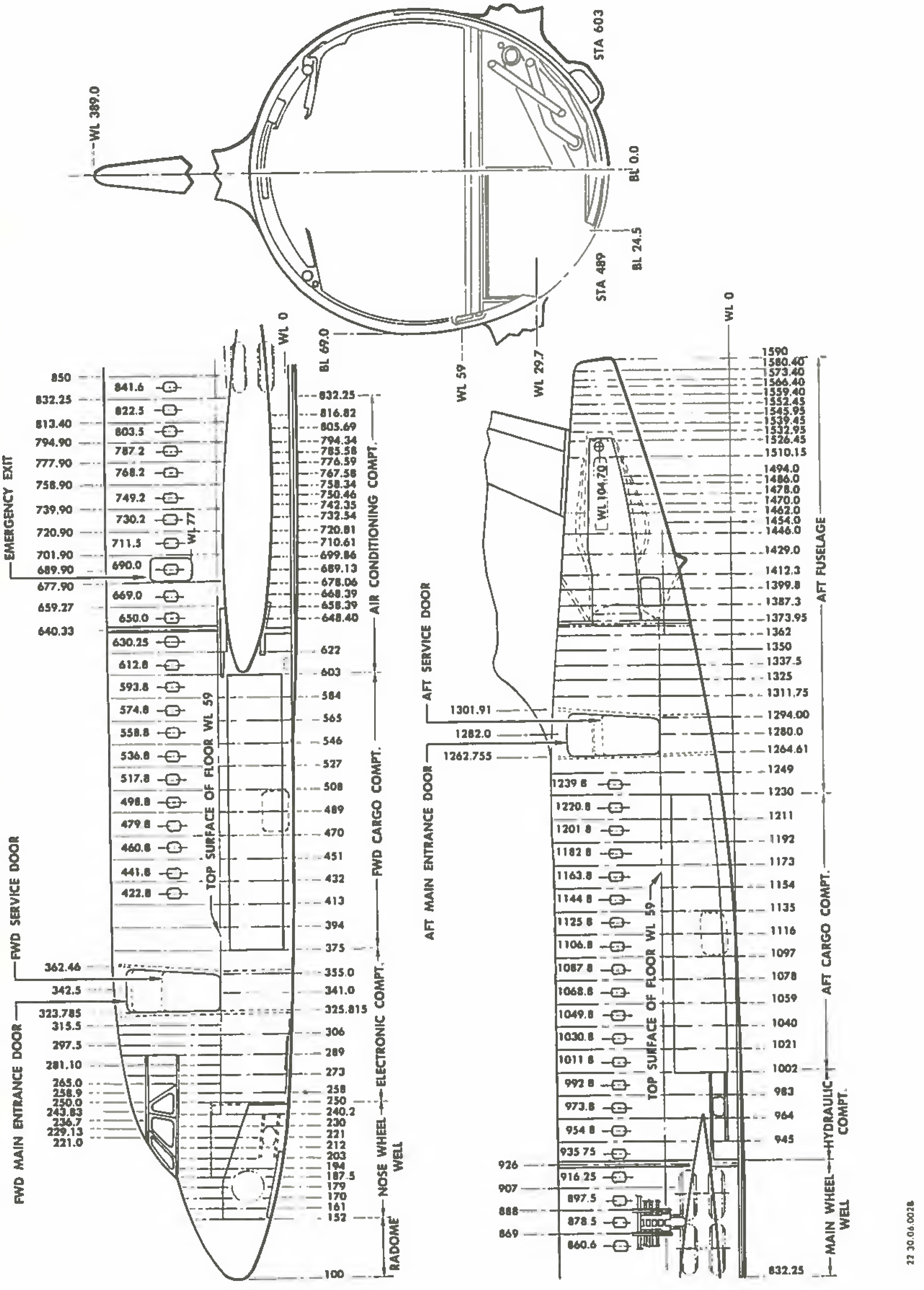
Station lines are unit measurements along the longitudinal axis of the fuselage, the lateral axis of the wings and horizontal stabilizer, and the vertical axis of the vertical stabilizer. All station lines are identified by their respective components, such as fuselage station lines, wing station lines, etc., and are always measured in inches.

Fuselage Station Lines

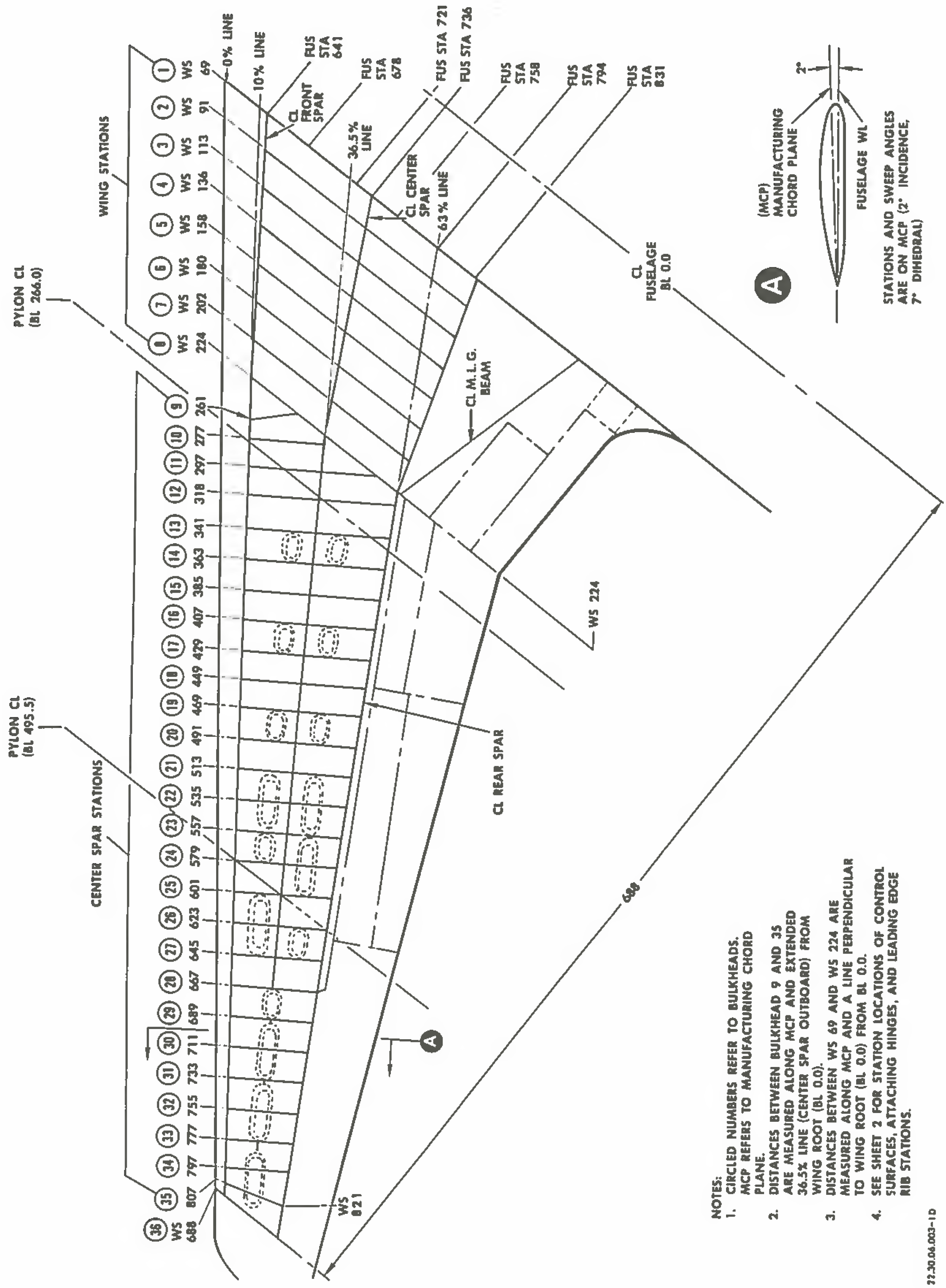
The foremost station on the fuselage nose is station 100, thus eliminating any negative or minus stations. Fuselage stations progress aft to the tail at station 1590. Figure 6-11 illustrates all the fuselage station lines for the frames, bulkheads, doors, windows, etc., from the tip of the radome to the tail.

Wing Station Lines

Wing stations are measured in inches from BL 0.0 to provide spanwise, or lateral axis, reference points and are illustrated in Figure 6-12. All wing stations from 0 to 69.518 and at station 688.254 are parallel to the airplane centerline (BL 0.0), while stations 261.413 through 797.540 are perpendicular to the wing center spar. Wing station 69.518 intersects the tailing edge at the break point and is a very useful reference point. In addition to the wing inch-stations, all wing bulkheads and/or ribs are numbered consecutively from 1 through 36 starting at the first wing bulkhead at wing station 69.518



Fuselage Station Lines
Figure 6-11



NOTES:
1. CIRCLED NUMBERS REFER TO BULKHEADS. MCP REFERS TO MANUFACTURING CHORD PLANE.
2. DISTANCES BETWEEN BULKHEAD 9 AND 35 ARE MEASURED ALONG MCP AND EXTENDED 36.5% LINE (CENTER SPAR OUTBOARD) FROM WING ROOT (BL 0.0).
3. DISTANCES BETWEEN WS 69 AND WS 224 ARE MEASURED ALONG MCP AND A LINE PERPENDICULAR TO WING ROOT (BL 0.0) FROM BL 0.0.
4. SEE SHEET 2 FOR STATION LOCATIONS OF CONTROL SURFACES, ATTACHING HINGES, AND LEADING EDGE RIB STATIONS.

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Wing Station Lines
Figure 6-12

outboard of the fuselage. Any of these numbers can be cross-referenced to a wing station if necessary. If needed, wing stations can be further identified as right-hand or left-hand.

Engine Pod Station Lines

The station numbering system for the engine pods and pylons is independent of all other station numbering systems (see Figure 6-13). A zero reference point, located 38.85 inches forward of the engine intake cowling on the longitudinal reference axis, has been utilized to determine the station measurements. The rear engine mount centerline is a centrally located point to use for reference during maintenance and it's station number is 200.0. Fuselage station lines, buttock lines and waterlines are also utilized for the location and installation of equipment.

Horizontal Stabilizer Station Lines

Horizontal stabilizer inch-stations are measured from a zero reference point and are perpendicular to the stabilizer front spar (see Figure 6-14). Buttock lines and fuselage station lines can also be used for location and installation of equipment in the horizontal stabilizer.

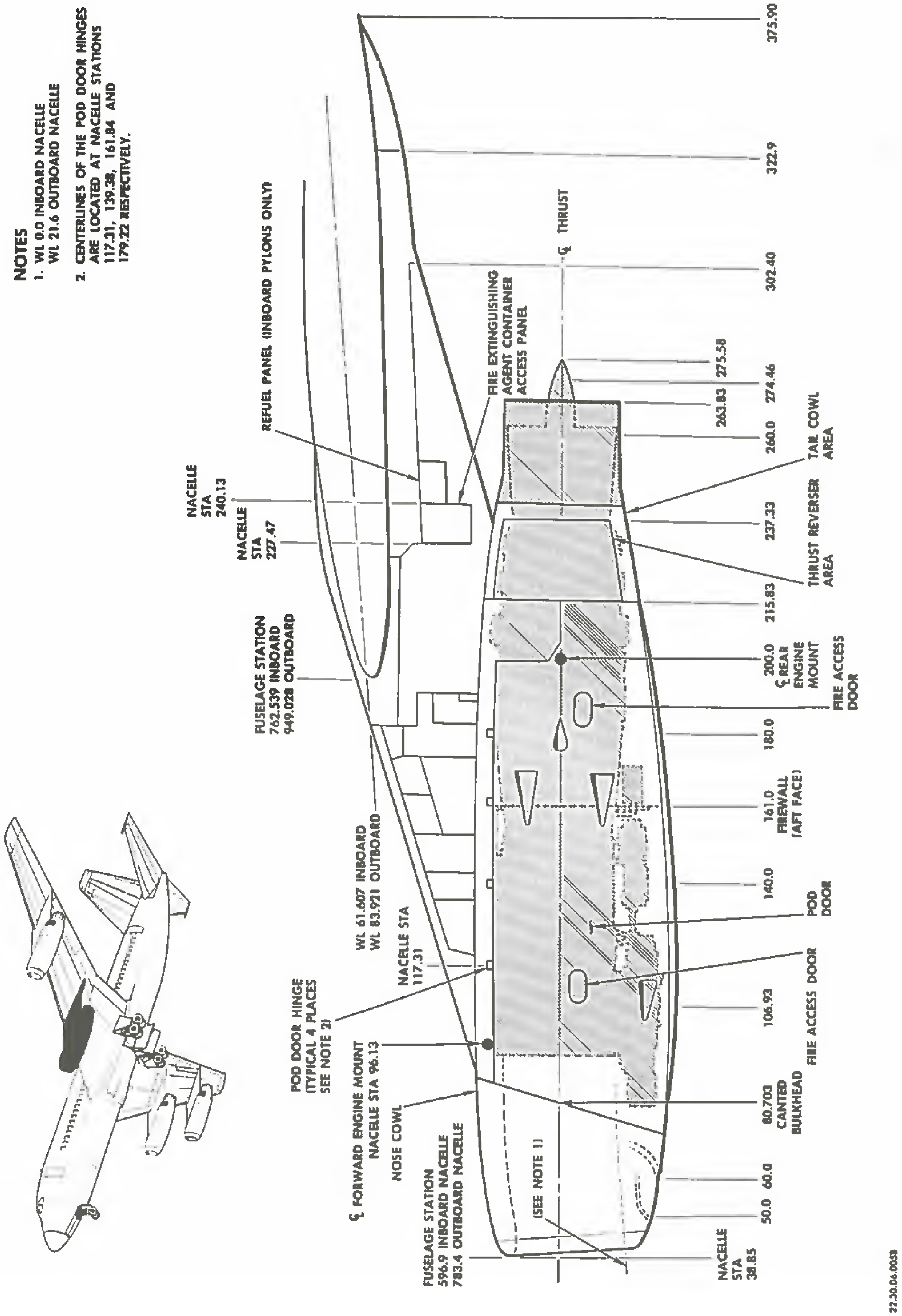
Vertical Stabilizer Station Lines

The vertical stabilizer inch-stations are measured from zero reference point and are perpendicular to the stabilizer front spar as shown on Figure 6-15. In addition to these stations, waterlines and fuselage station lines can be utilized for installation and location of equipment.

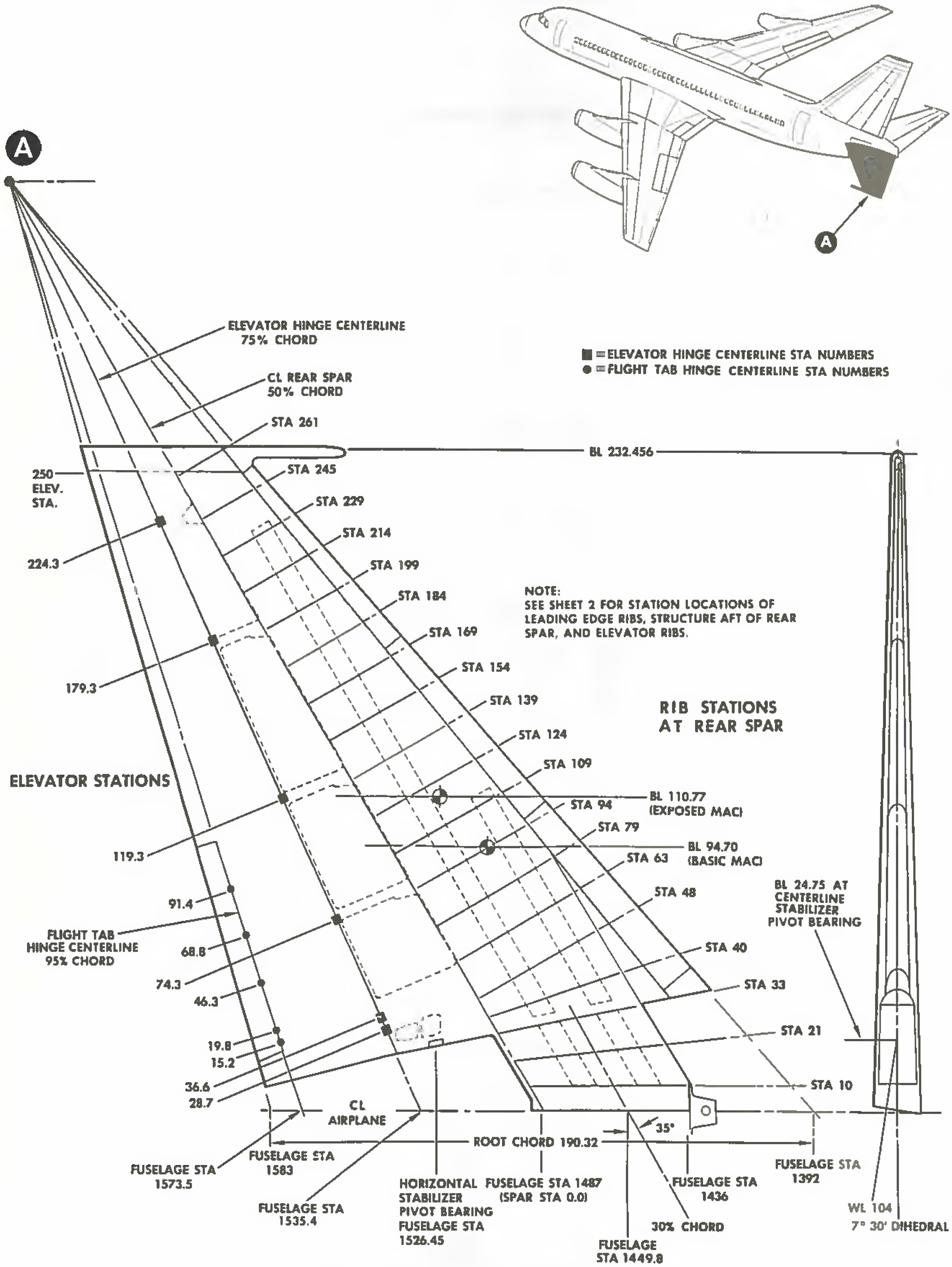
Water and Buttock Station Lines

Waterlines are vertical reference points (WL) measured in inches from the inside surface of the lower fuselage skin at the airplane centerline. This inside surface of the lower fuselage skin is at WL 0.0; all waterlines below this point are negative, or minus, waterlines. Several easy waterlines to remember are: WL 0.0 at the bottom of the fuselage centerline, the top surface of the cabin floor at WL 59.00, and the fuselage leveling rivets (6) on the outside surface of the fuselage (right and left sides) at WL 68.00 (see Figure 6-16).

Buttock lines (BL) are reference points along the lateral or Y-axis and are measured in inches to the right and left of the fuselage centerline, also illustrated in Figure 6-16. The fuselage centerline is at BL 0.0, while the inside surface of the fuselage skin at the widest point in the fuselage (WL 81.75) is at BL 69.00. Buttock lines are also used as reference points in the wings and horizontal stabilizer, and extend to the right and left of the fuselage centerline. The extreme side of the wing tips are at BL 720.00.



Pod and Pylon Station Lines
Figure 6-13



HORIZONTAL STABILIZER DIAGRAM

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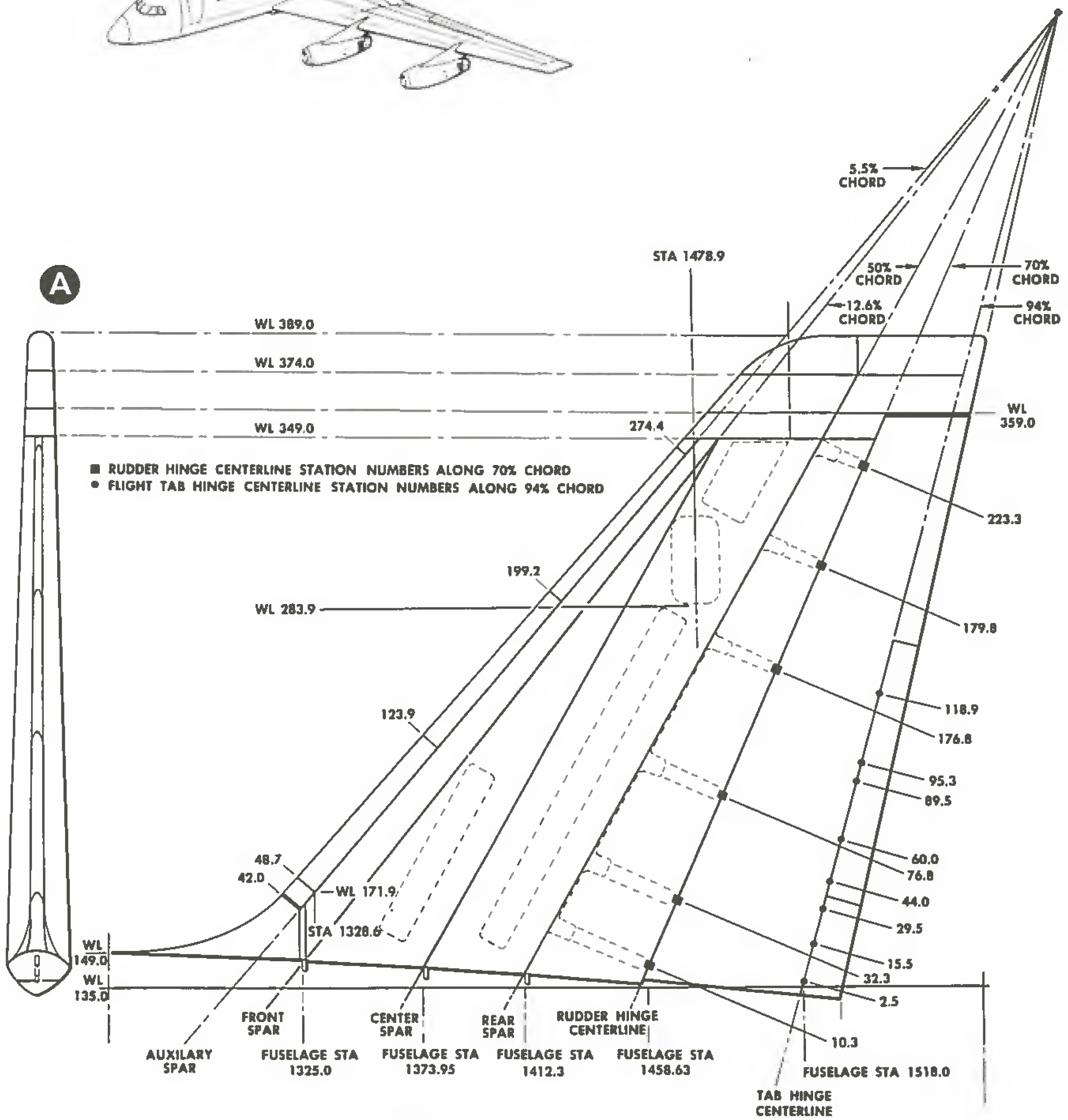
Horizontal Stabilizer Station Lines
Figure 6-14

NOTE:
SEE SHEET 2 FOR STATION LOCATIONS OF THE LEADING EDGE RIBS, INTERSPAR RIBS, STRUCTURE AFT OF REAR SPAR, AND RUDDER RIBS.



VERTICAL STABILIZER STATION DIAGRAM

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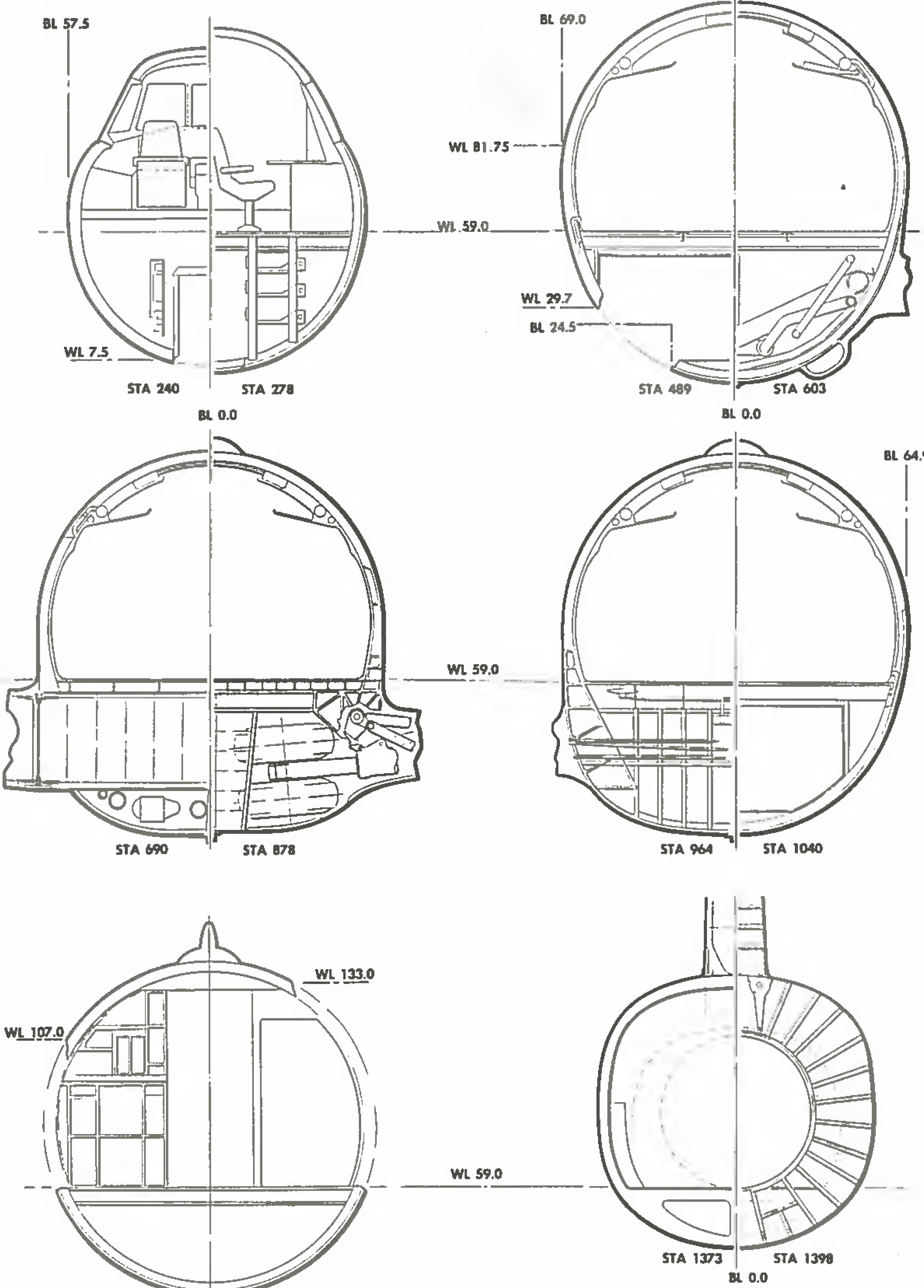
Vertical Stabilizer Station Lines
Figure 6-15



CROSS SECTIONS—TYPICAL

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Dimensions

Wing

Span	120.00 feet
Chord at Root	35.70 feet
Chord at Tip	6.76 feet

Landing Gear

Wheel base	53.006 feet
Tread (Main Gears)	18.880 feet
Tread (Main Gear Tires)	
Longitudinal Centerline	45.00 inches
Spanwise Centerline	21.50 inches
Tread (Nose Gear Tires)	17.00 inches
Nose Gear Centerline to Station 100	11.77 feet

Areas

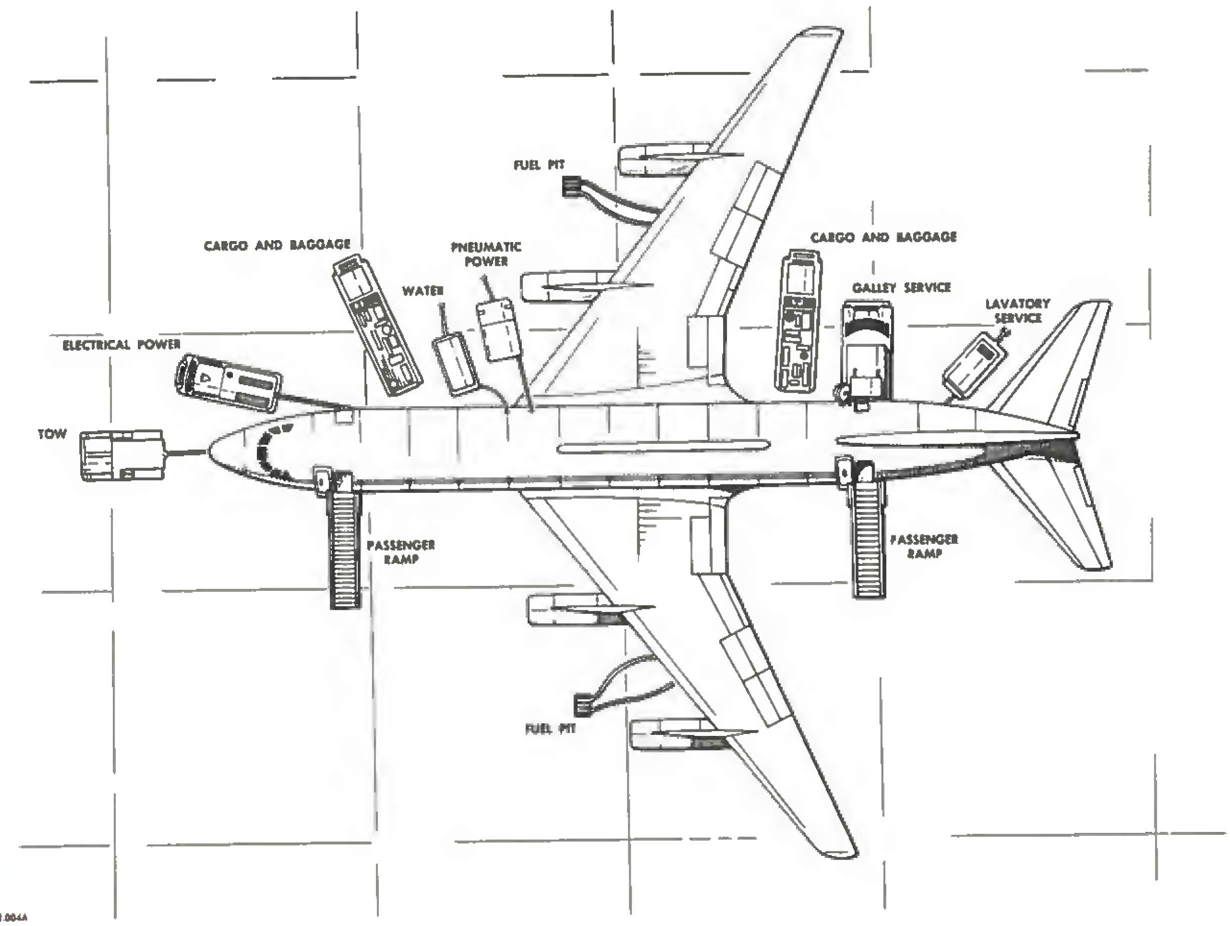
Wing (Basic)	2000.0 square feet
Vertical Stabilizer (Total)	375.0 square feet
Rudder (Aft of Hinge Line, Including Tabs)	85.5 square feet
Rudder Balance	45.3 square feet
Servo Tab (Aft of Hinge)	7.9 square feet
Trim Tab (Aft of Hinge)	3.8 square feet
Horizontal Stabilizer (Total)	395.0 square feet
Elevator (Aft of Hinge, Including Tabs)	92.0 square feet
Tab (Aft of Hinge)	6.6 square feet
Lower Aft Cargo Compartment	415 cubic feet
Forward Cargo Compartment	450 cubic feet

GENERAL SERVICING DATA

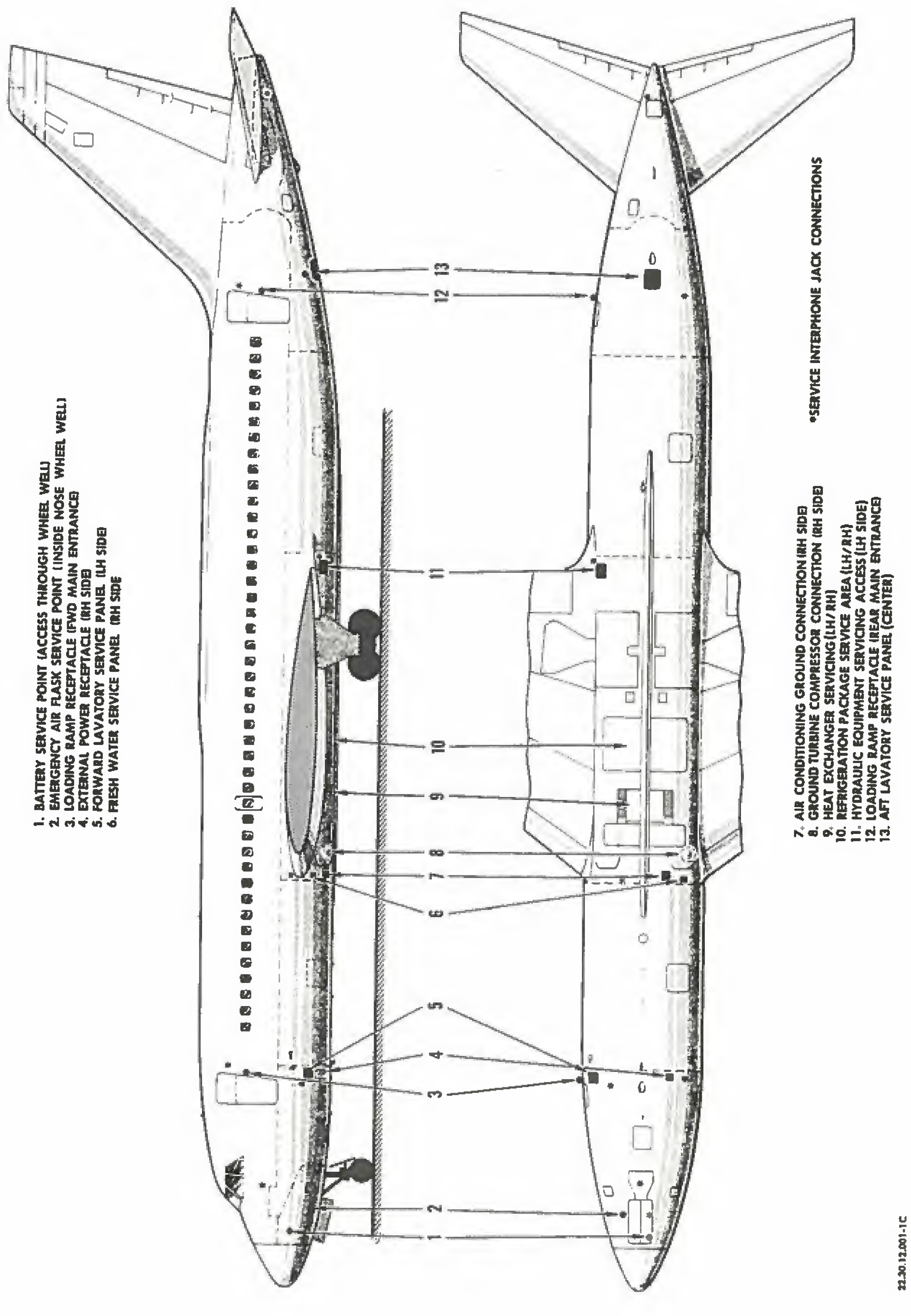
The Convair 880 is designed so that it can be easily serviced in a minimum of time with little interference between the crews servicing the airplane systems, galleys, lavatories and loading and unloading the cargo areas. All servicing of the fuselage systems and compartments is accomplished from the right side of the airplane; the left side of the airplane, except for fuel and engine servicing, is reserved for passenger and flight crew enplaning and deplaning (see Figure 6-17). Various servicing points and locations are shown in Figure 6-18.

Servicing Replenishment Chart

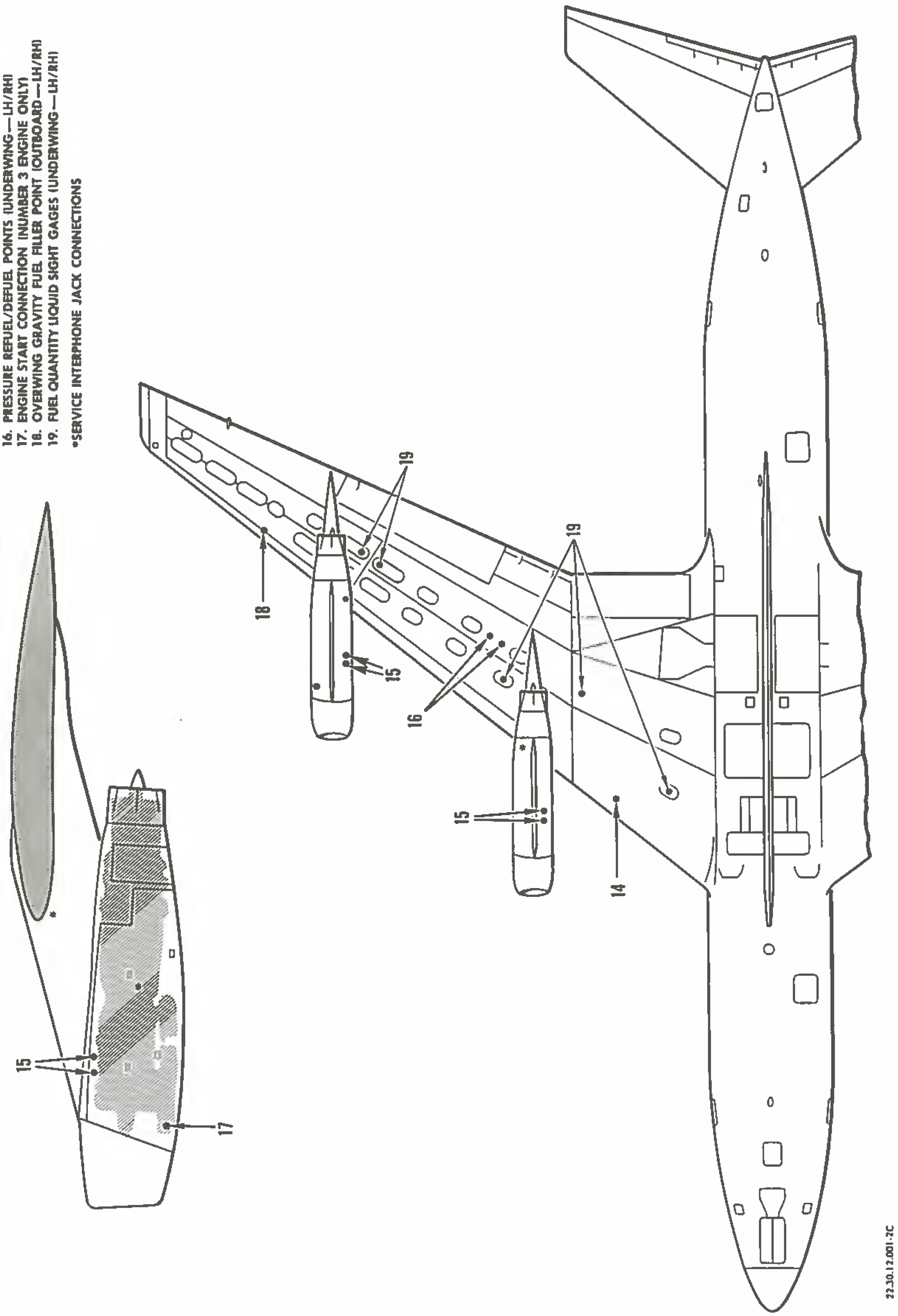
SYSTEM OR TANK	USABLE CAPACITY U. S. GALLONS	IMPERIAL GALLONS	LITERS
Fuel Tanks (JP-4)			
No. 1 Main and Replenishing	2,362.00	1,960.46	8,940.17
No. 2 Main and Replenishing	2,930.00	2,431.90	11,090.05
No. 3 Main and Replenishing	2,930.00	2,431.90	11,090.05
No. 4 Main and Replenishing	2,362.00	1,960.46	8,940.17
TOTALS	10,584.00	8,784.72	40,060.44
CSD and Engine Oil Tanks (Specification MIL-L-7808C)			
No. 1 (CSD-1.72, ENG. -4.15)	5.87	4.88	22.21
No. 2 (Same)	5.87	4.88	22.21
No. 3 (Same)	5.87	4.88	22.21
No. 4 (Same)	5.87	4.88	22.21
TOTALS	23.48	19.52	88.84



Airplane Ground Servicing - Typical
Figure 6-17



Airplane Servicing Points
Figure 6-18 (Sheet 1 of 2)



14. OVERWING GRAVITY FUEL FILLER POINT (INBOARD—LH/RH)
15. ENGINE AND CSD OIL FILLER POINTS (EACH NACELLE) (INBD SIDE NO. 1 AND 2 POD, OUTBD SIDE NO. 3 AND 4 POD)
16. PRESSURE REFUEL/DEFUEL POINTS (UNDERWING—LH/RH)
17. ENGINE START CONNECTION (NUMBER 3 ENGINE ONLY)
18. OVERWING GRAVITY FUEL FILLER POINT (OUTBOARD—LH/RH)
19. FUEL QUANTITY LIQUID SIGHT GAGES (UNDERWING—LH/RH)

*SERVICE INTERPHONE JACK CONNECTIONS

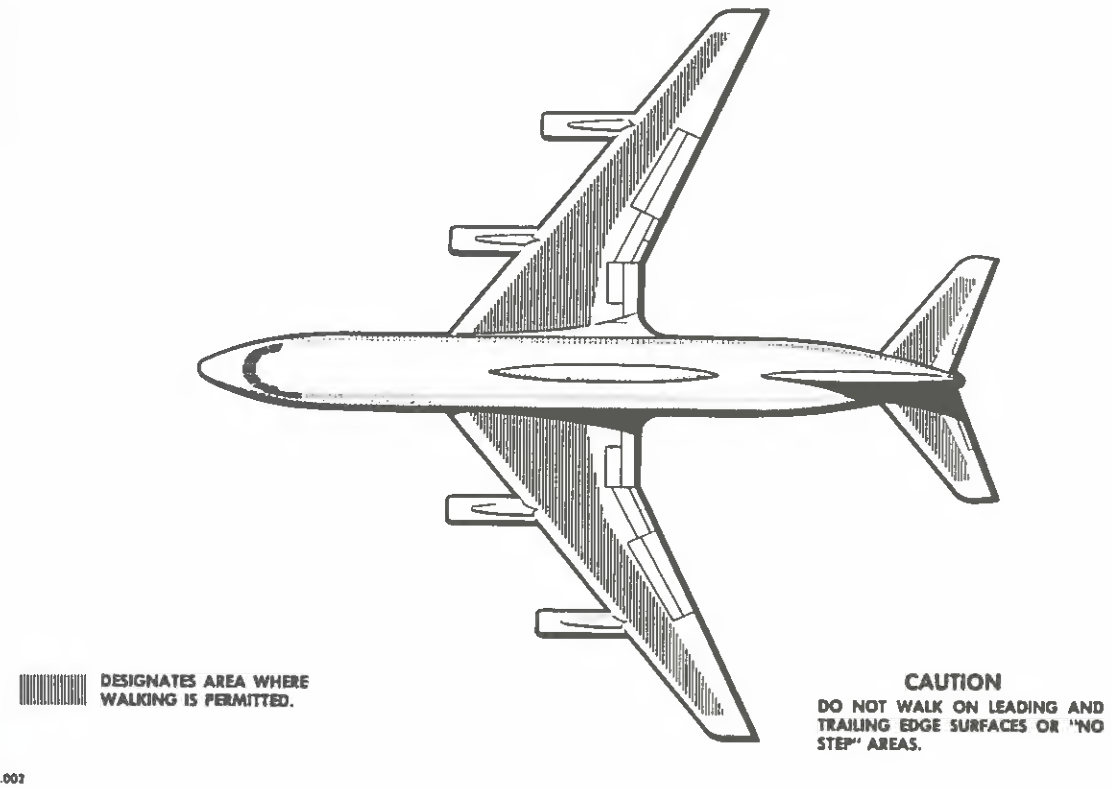
Airplane Servicing Points
Figure 6-18 (Sheet 2 of 2)

Servicing Replenishment Chart (Cont)

SYSTEM OR TANK	USABLE CAPACITY U. S. GALLONS	IMPERIAL GALLONS	LITERS
Fresh Water Tank (Main Supply)	50.00	41.63	189.27
Main Hydraulic Supply (SKYDROL-500)			
No. 1 Reservoir	2.60	2.16	9.84
No. 1 System (less reservoir)	11.40	9.50	43.14
No. 2 Reservoir	8.90	7.45	33.68
No. 2 System (less reservoir)	22.10	18.40	83.65
TOTALS	45.00	37.51	170.31

Walkways

Walkways, as such, are not designated since this airplane has been designed to permit as much servicing as possible to be accomplished either from work stands or from the ground. Figure 6-19 illustrates that the area between the front and rear spars on the wings and the horizontal stabilizer are the only places where walking is permitted. The areas on these surfaces on which walking is not permitted are marked "NO STEP". Maintenance personnel should remove their shoes or wear protective slip-on shoe coverings any time it is necessary to walk on the wings or horizontal stabilizer so as to preserve the required aerodynamic smoothness of these surfaces.



Walkways Diagram
Figure 6-19

AIRPLANE TOWING PROCEDURE

Towing is normally accomplished from the nose gear when the airplane is on a hard, dry surface. When mud, snow or rough terrain may create excessive towing loads, towing must be accomplished from the main gears. Main gear towing can be either backward or forward as required by circumstances.

NOTE:
1. Place down-lock safety pins in main and nose gear braces.
2. Make an operational check of the hydraulic brake system.
3. Check emergency air brake pressure.
4. Disconnect the nose wheel torque arms.

Nose Gear Towing

Equipment required for nose gear towing consists of a tow bar and a tow truck equipped with an electrical power source suitable for use with the airplane. Four men should be used during towing activities; one for the operation of wheel brakes and equipment in the cockpit, one on the tug, and one at each wing tip to check wing and tail clearance. Towing speed should not exceed a fast walking gait (approximately 5 to 6 mph). The nose wheel steering disconnect permits a swivel of 360 degrees; however, towing at angles greater than 50 degrees can result in scuffing of the main tires (see Figures 6-20 and 6-21).

CAUTION: DO NOT DISCONNECT EQUIPMENT OR LINES AT THE NOSE GEAR. IMPROPER RE-CONNECTION CAN CAUSE MALFUNCTION OF ONE OR MORE SYSTEMS.

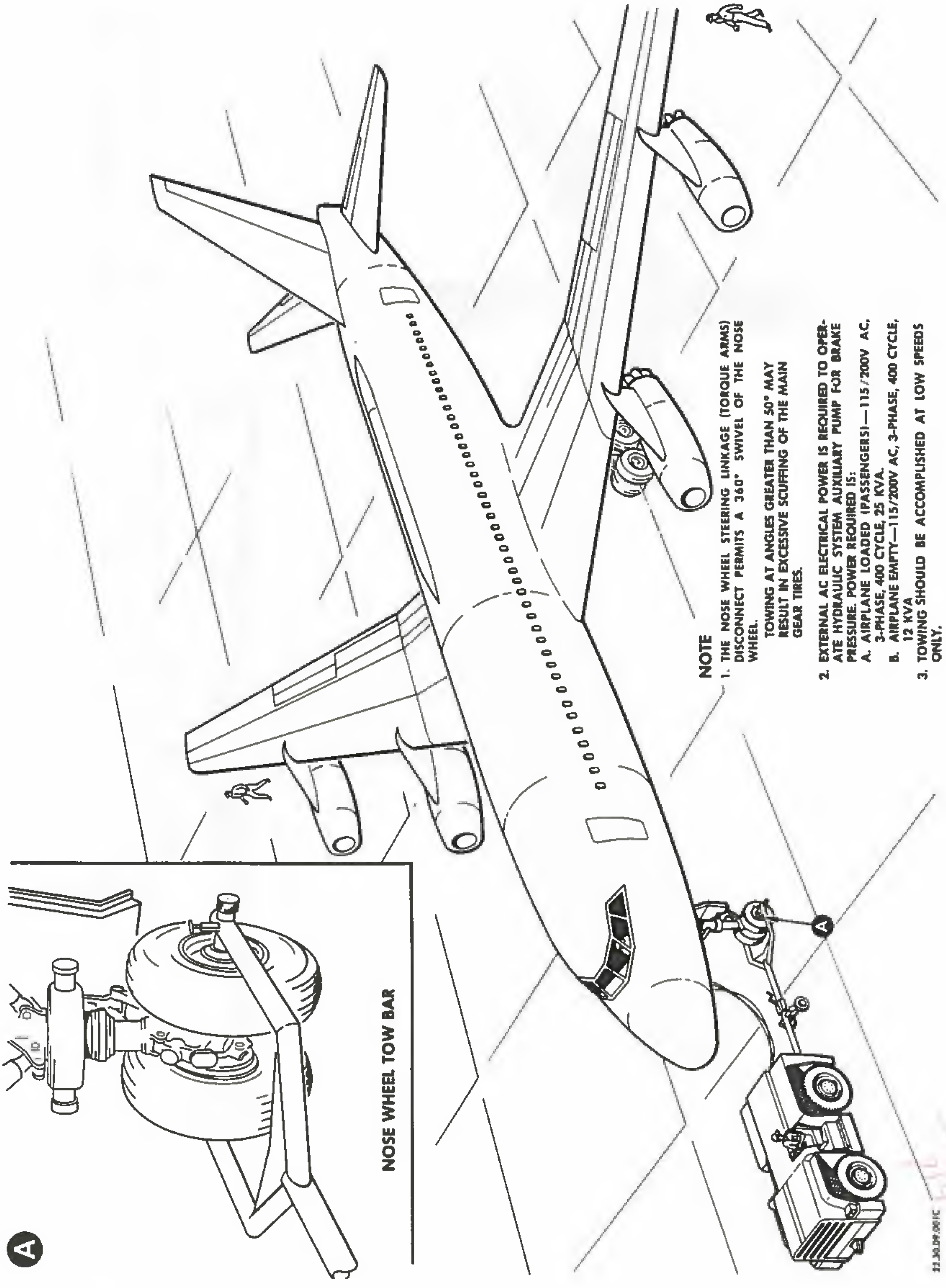
Main Gear Towing

Integral towing lugs are installed on the front and rear of each main gear truck (see Figure 6-22). By utilizing a manila rope or aircraft cable, towing can be accomplished either forward or backward in mud, snow, or over rough terrain. Personnel should be utilized in the same manner as for nose gear towing. Brake and electrical requirements are also the same.

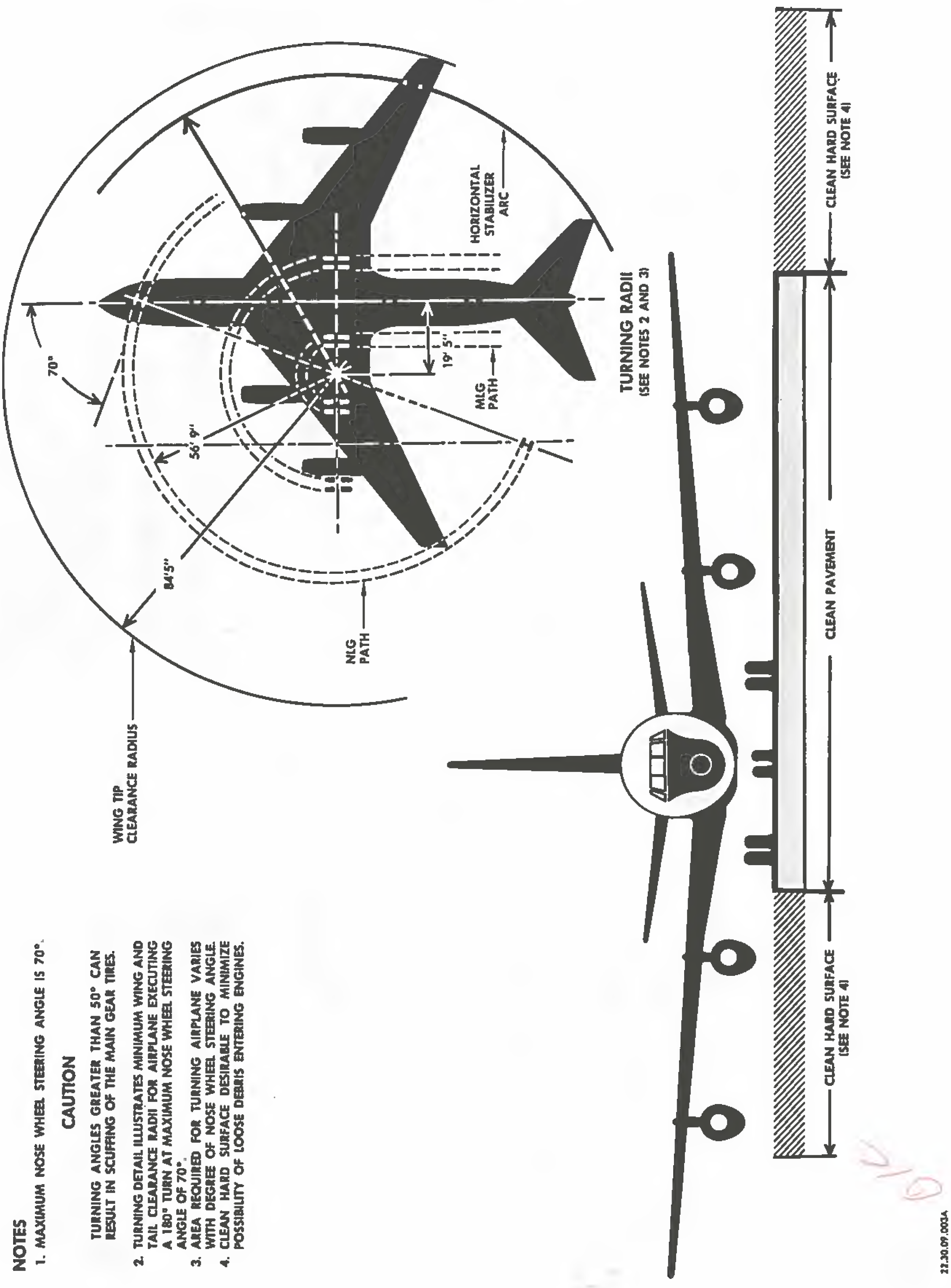
AIRPLANE JACKING PROCEDURES

Lifting the airplane consists of such operations as jacking the entire airplane, jacking at each landing gear, hoisting the airplane, and raising the airplane with pneumatic lifting bags. Precautions such as removing all unnecessary equipment in the area and the checking of overhead clearances should be observed.

WARNING: DURING JACKING AND LIFTING PROCEDURES, ALL PERSONNEL NOT ACTIVELY ENGAGED IN THE OPERATION SHOULD REMAIN CLEAR OF THE AIRPLANE. ALL EQUIPMENT SHOULD BE REMOVED FROM THE AREAS DIRECTLY BENEATH FUSELAGE, WINGS AND TAIL.



Towing From The Nose Gear
Figure 6-20



NOTES

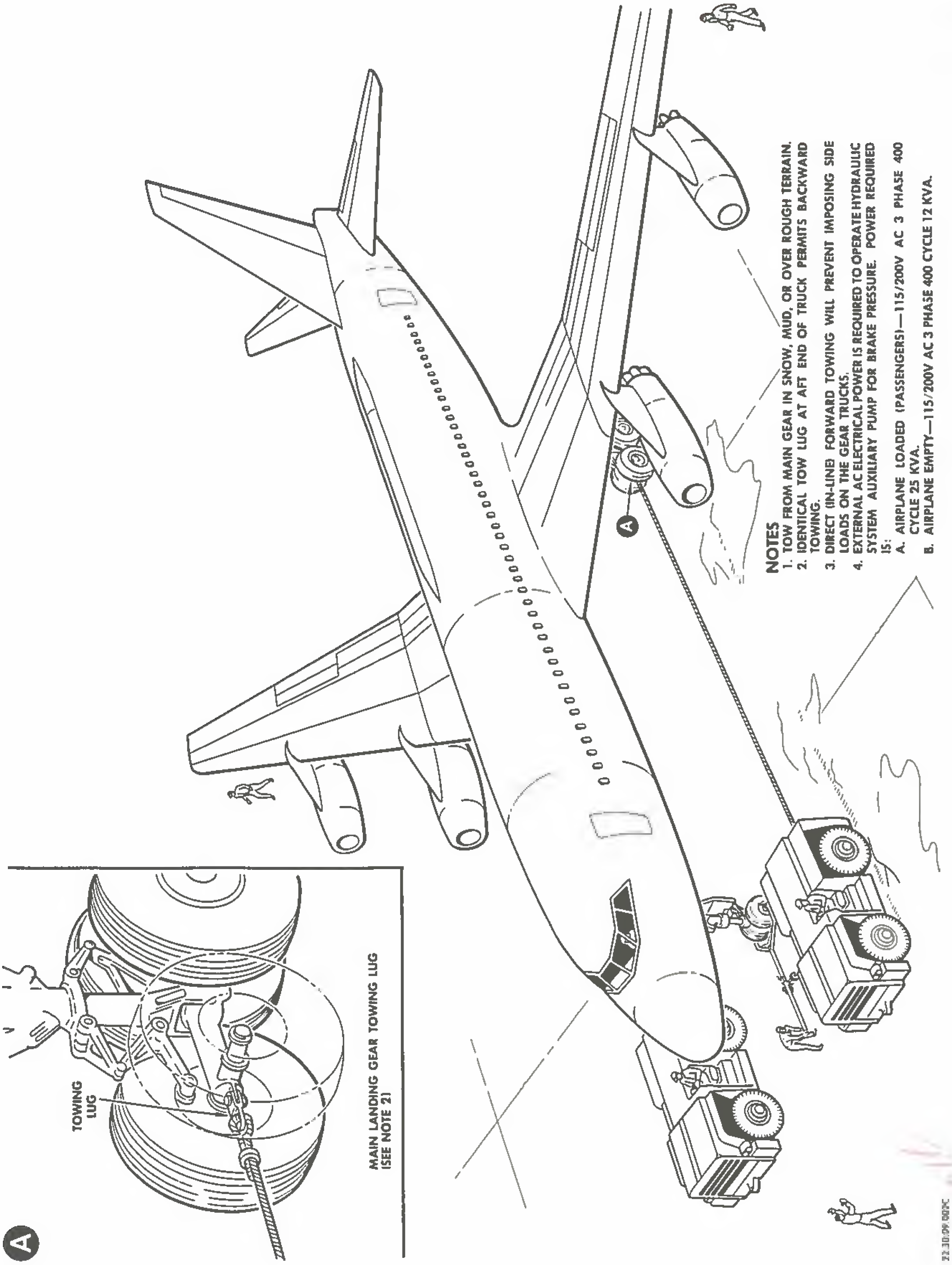
1. MAXIMUM NOSE WHEEL STEERING ANGLE IS 70°.

CAUTION

TURNING ANGLES GREATER THAN 50° CAN RESULT IN SCUFFING OF THE MAIN GEAR TIRES.

2. TURNING DETAIL ILLUSTRATES MINIMUM WING AND TAIL CLEARANCE RADII FOR AIRPLANE EXECUTING A 180° TURN AT MAXIMUM NOSE WHEEL STEERING ANGLE OF 70°.
3. AREA REQUIRED FOR TURNING AIRPLANE VARIES WITH DEGREE OF NOSE WHEEL STEERING ANGLE.
4. CLEAN HARD SURFACE DESIRABLE TO MINIMIZE POSSIBILITY OF LOOSE DEBRIS ENTERING ENGINES.

Airplane Turning Radii
Figure 6-21



NOTES

1. TOW FROM MAIN GEAR IN SNOW, MUD, OR OVER ROUGH TERRAIN.
2. IDENTICAL TOW LUG AT AFT END OF TRUCK PERMITS BACKWARD TOWING.
3. DIRECT (IN-LINE) FORWARD TOWING WILL PREVENT IMPOSING SIDE LOADS ON THE GEAR TRUCKS.
4. EXTERNAL AC ELECTRICAL POWER IS REQUIRED TO OPERATE HYDRAULIC SYSTEM AUXILIARY PUMP FOR BRAKE PRESSURE. POWER REQUIRED IS:
   A. AIRPLANE LOADED (PASSENGERS)—115/200V AC 3 PHASE 400 CYCLE 25 KVA.
   B. AIRPLANE EMPTY—115/200V AC 3 PHASE 400 CYCLE 12 KVA.

Towing From The Main Gear
Figure 6-22

Jacking the Entire Airplane

The entire airplane can be raised using the three airplane jack points (see Figure 6-23). The fuselage jack point is located at station 153.6 just forward of the nose landing gear well. The two wing jack points are located at wing station 224.7 four inches aft of the rear spar. When not in use, the jack fitting openings are covered with flush covers to retain aerodynamic smoothness.

CAUTION: 1. JACKING OPERATIONS MUST BE SIMULTANEOUS, WITH THE AIRPLANE KEPT IN A LEVEL ATTITUDE.
2. PLACE A RESTRAINING CLAMP ON EACH GEAR STRUT TO PREVENT STRUT EXTENSION WHEN LIFTING AIRPLANE.

Jacking at the Landing Gear

Three integral jack pads are installed on the main gear trucks and one on the nose gear strut (see Figure 6-24). Individual wheel or brake servicing is accomplished by raising the front or rear of the main gear trucks by means of the integral pads. An added safety precaution when jacking under a strut is to place a jack under the respective wing or fuselage jack point. When jacking under the strut, only lift the gear high enough so that the tires clear the ground.

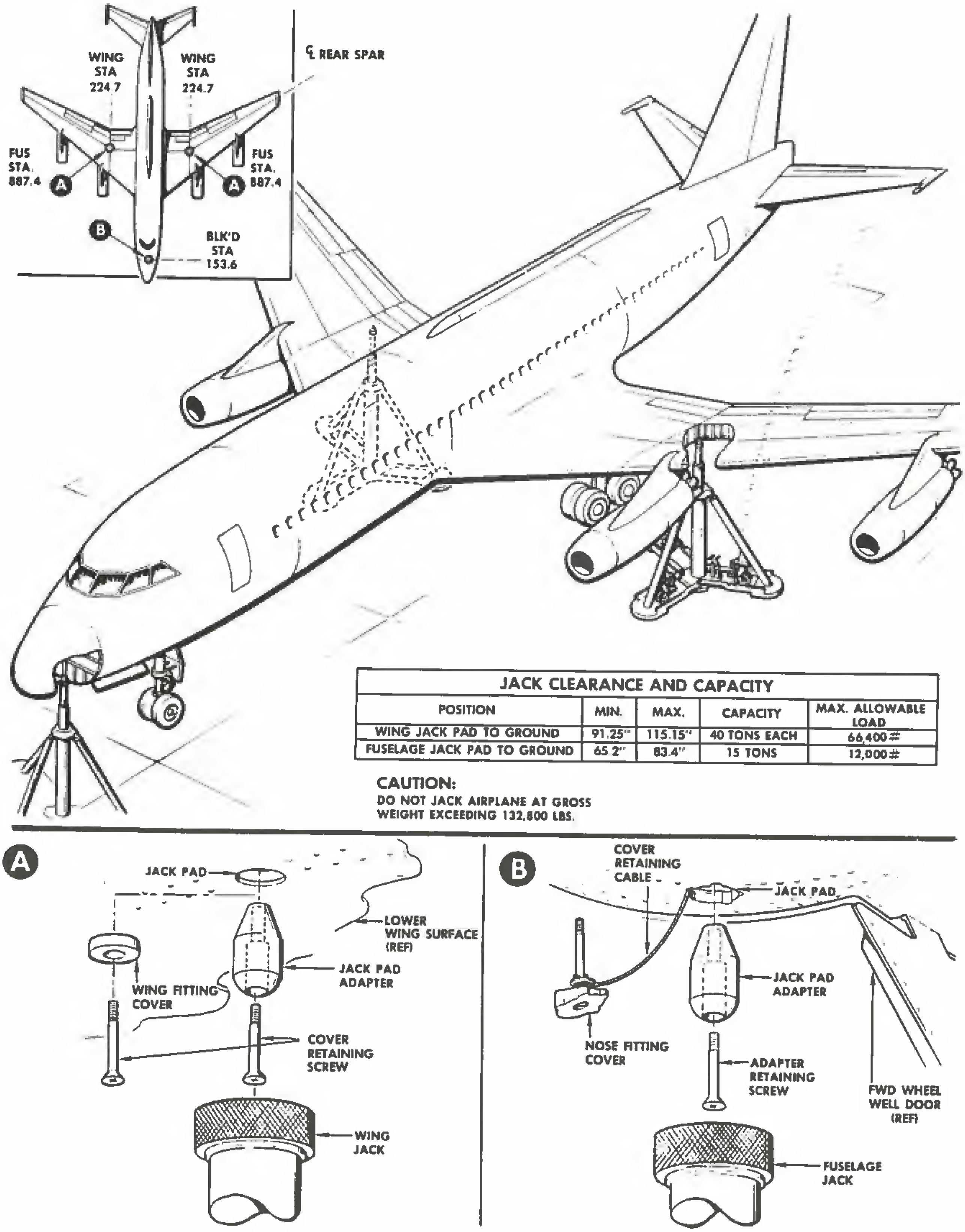
Lifting with Pneumatic Bags

In case of soft terrain or wheels-up landing, the airplane can be lifted by using pneumatic bags. Place the bags under structural points of the wings and fuselage and inflate evenly, holding the airplane as level as possible. Sharp objects on the airplane surfaces or on the ground must be removed to prevent puncturing the bags.

CAUTION: MOOR THE AIRPLANE LATERALLY AND LONGITUDINALLY BEFORE INFLATING BAGS. ALL PERSONNEL AND EQUIPMENT NOT ACTIVELY ENGAGED IN THE OPERATION MUST BE REMOVED TO A SAFE DISTANCE.

AIRPLANE MOORING PROVISIONS

Mooring provisions are provided on the forward and aft main gear trucks and on the tail skid (see Figure 6-25). Tie-down is accomplished by utilizing manila rope or aircraft cable between the mooring lugs and the tie-down points on the ramp. Integral control surface gust dampers are provided in the flight control system to protect the airplane control surfaces from wind damage. These dampers are constantly engaged and do not interfere with control surface movement.



JACK CLEARANCE AND CAPACITY

POSITION	MIN.	MAX.	CAPACITY	MAX. ALLOWABLE LOAD
WING JACK PAD TO GROUND	91.25"	115.15"	40 TONS EACH	66,400#
FUSELAGE JACK PAD TO GROUND	65.2"	83.4"	15 TONS	12,000#

CAUTION:
DO NOT JACK AIRPLANE AT GROSS WEIGHT EXCEEDING 132,800 LBS.

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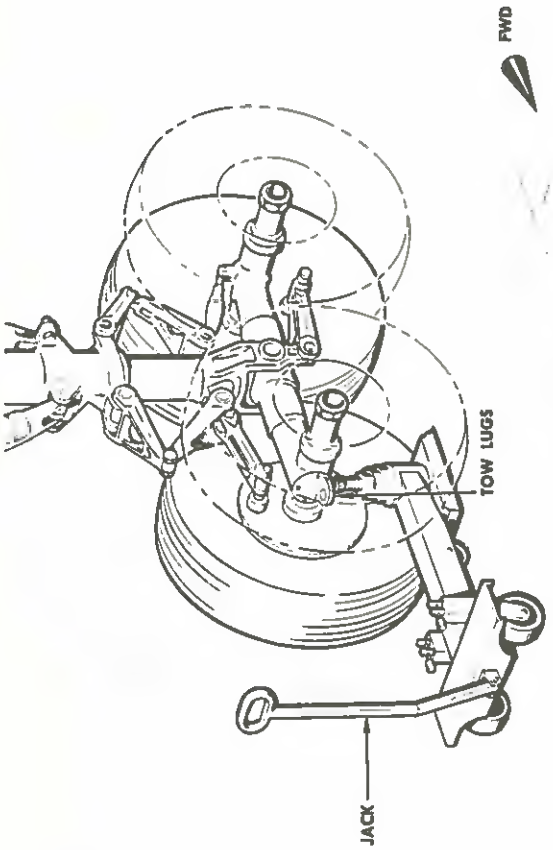
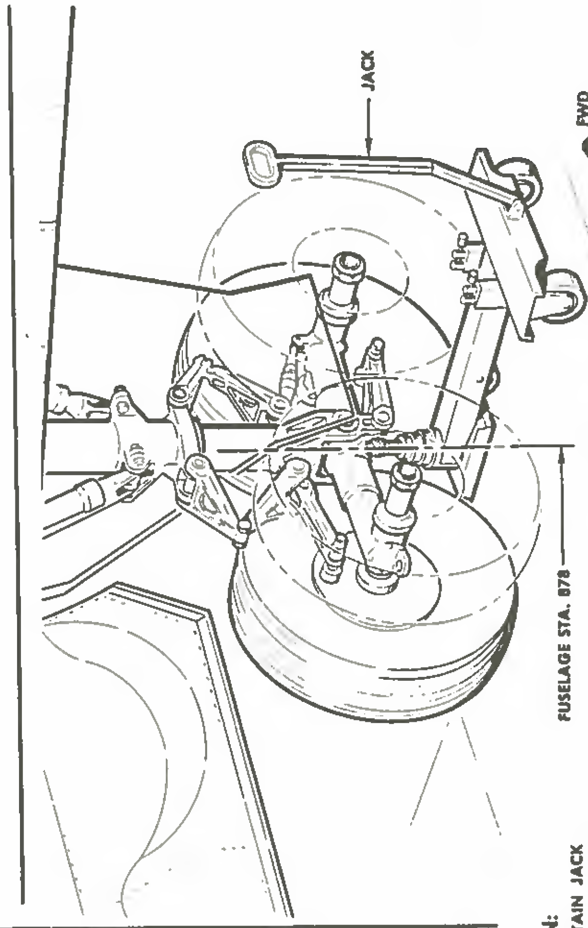
Airplane Jack Points
Figure 6-23



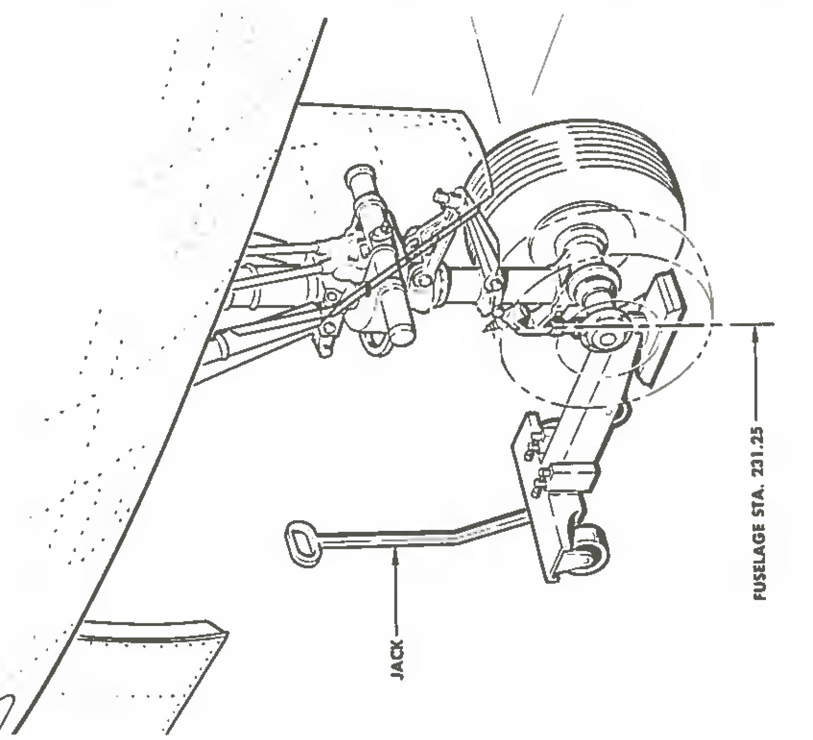
JACKING AT
MAIN LANDING GEAR TRUCK



JACKING AT
MAIN LANDING GEAR STRUT

JACK PAD TO SURFACE DIMENSIONS

JACK POSITION	TIRES INFLATED	4 TIRES FLAT	2 TIRES FLAT	REQUIRED JACK CAPACITY	MAXIMUM ALLOWABLE LOAD
MLG STRUT JACK POINT	11.62"	6.95"	9.88"	50 TONS	86,700 #
FORE OR AFT TRUCK JACK POINT	12.62"		6.95"	25 TONS	FWD 45,200 # AFT 41,500 #
NLG	13.8"		11.1"	25 TONS	15,000 #



JACKING AT
NOSE LANDING GEAR

CAUTION:
MAKE CERTAIN JACK IS CENTERED ON JACK PAD BEFORE LIFTING LANDING GEAR.

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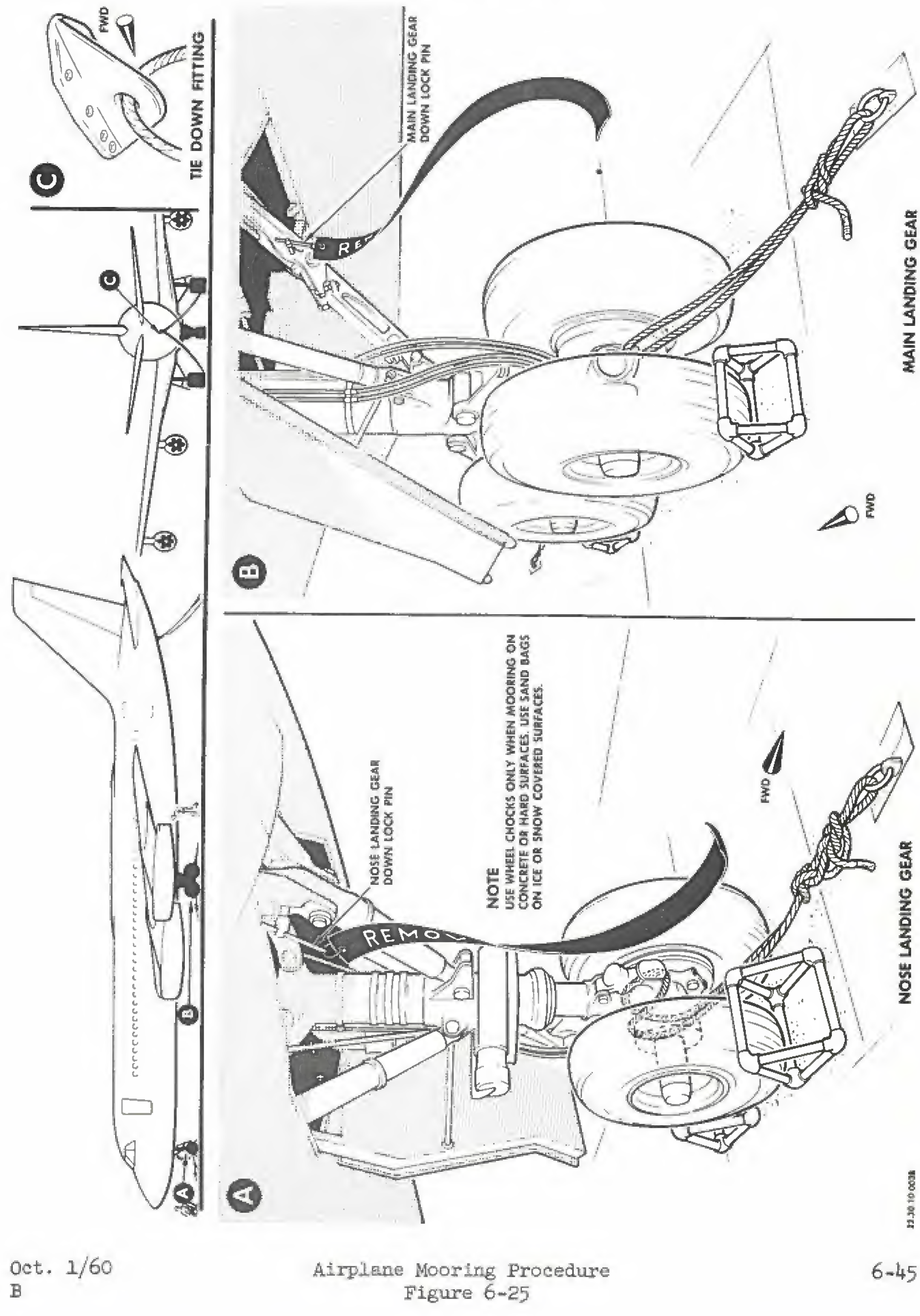
Airplane Landing Gear Jack Points
Figure 6-24



Airplane Mooring Procedure
Figure 6-25

AIRPLANE PARKING

When parking the airplane for a short time, the parking brake should be set by depressing the brake pedals and pulling the parking brake handle on. Always install landing gear down-locks, chocks and static ground cables. When unfavorable weather prevails, care should be taken to shield the airplane as much as possible and head it into the wind. During windy and dusty weather, window and door openings should be closed and safety covers installed where necessary. Extreme winter weather conditions will require ice and snow shields over the windshields, wings and tail surfaces. Precautions should also be taken to prevent the airplane tires from freezing to the ramp. Parking brakes should NOT be set in cold weather as an accumulation of moisture may cause them to freeze. A parking brake warning light, actuated by a microswitch in the parking brake mechanism, is installed on the pilot's auxiliary instrument panel. The light will illuminate when the brake is on.

Section 7

FLIGHT CONTROLS

TABLE OF CONTENTS

GENERAL FLIGHT CONTROL SYSTEMS

The Convair 880 flight controls consist of four primary and six secondary flight control systems (see Figure 7-1). The primary systems include the rudder, elevator, aileron and spoiler controls. The secondary systems include all the primary control surface trim systems, the flaps, the speed brake function of the spoilers and the speed brake function of the main landing gear.

Primary Flight Controls

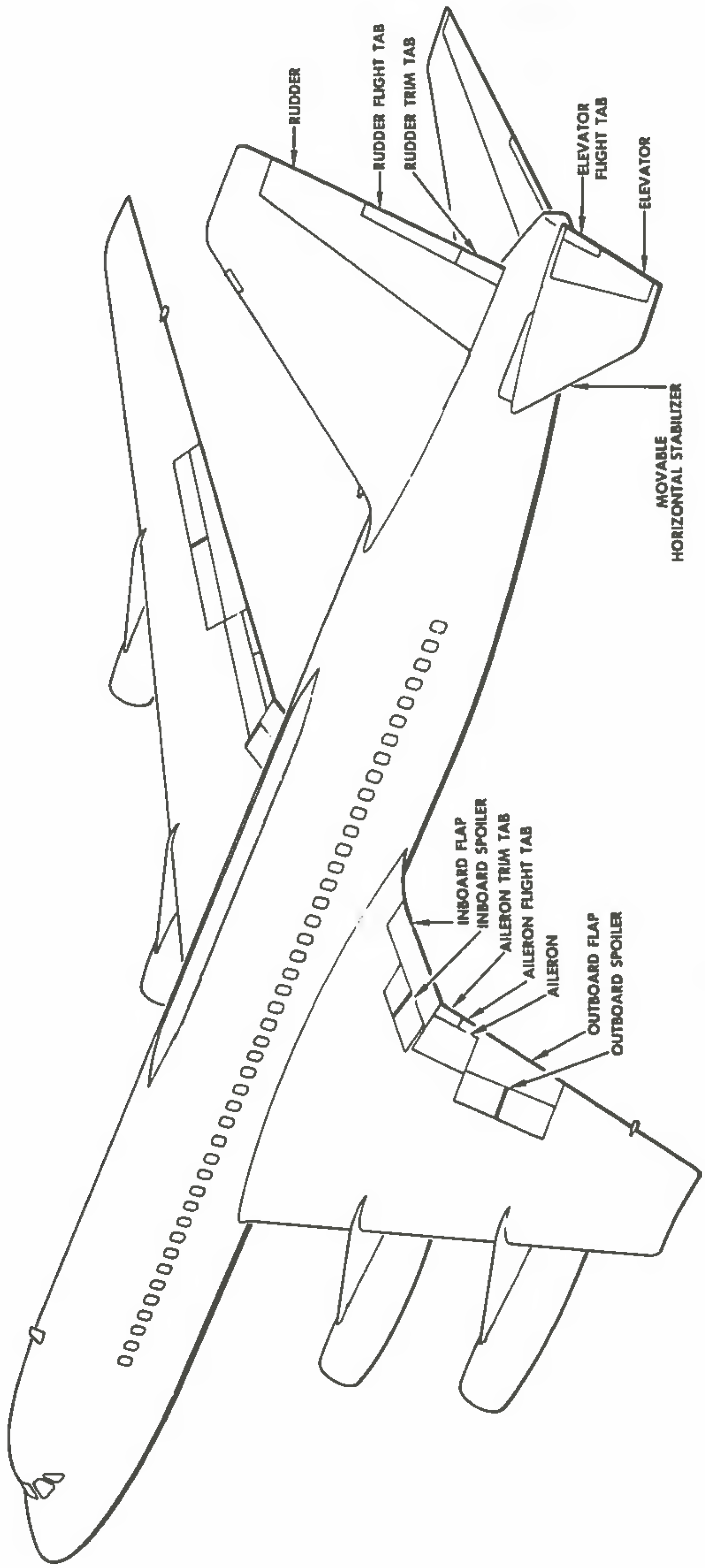
All primary flight control systems except the spoilers utilize a flight tab arrangement such that pilot action at the controls moves a slight tab, causing the main control surface to move in the desired direction. Due to the large mechanical advantage of the flight tab over the main control surface, plus the mechanical advantage of the pilot over the tab, the large forces required to move the main control surfaces are easily overcome by relatively small pilot input forces. Balance boards are used to reduce the main control surface hinge moment to minimize pilot input to the rudder, aileron and elevator control systems. The spoiler system performs functions in both the primary and secondary systems. The primary function is to provide the principal means of lateral control of the airplane. The spoilers are augmented by the aileron system.

Secondary Flight Controls

The aileron and rudder trim systems use conventional trim tabs positioned by moving control knobs in the flight compartment. Instead of elevator trim tabs, the horizontal stabilizer moves as a unit for airplane longitudinal trim. The stabilizer is mechanically controlled and hydraulically operated during normal operation. An emergency trim system can be operated either electrically or mechanically from the flight compartment. The hydraulically operated stabilizer trim system is also electrically controlled by a sliding button on each pilot's control wheel.

In the secondary control system, spoiler function is to act as an airplane speed brake. Moving a speed brake control handle in the flight compartment to the desired speed setting actuates the spoilers for symmetrical speed brake use. The inboard or outboard spoilers can be raised independently by means of electric actuator mechanisms to provide emergency longitudinal trim in case the stabilizer jams in the maximum airplane nose up or down position.

Double-slotted Fowler type flaps are provided on the wing trailing edge. Each wing incorporates two hydraulically operated flaps that are controlled from the flight compartment.



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Flight Control System
Figure 7-1

A ground gust protection system is included as part of the secondary flight control system to prevent wind damage to primary control surfaces. The system is a permanent installation requiring no manual control or selection, and does not interfere with normal control surface movement.

FLIGHT CONTROL HYDRAULIC SYSTEMS

Hydraulic pressure is used in the flight control system to operate the flaps, spoilers, and the horizontal stabilizer trim systems. Figures 7-2 and 7-3 show schematic diagrams of the flight control hydraulic systems.

Flap Hydraulic System

The flaps are moved to the desired position when the pilot moves the flap control lever on the pilots' pedestal to a selected position corresponding to the degree of flap extension desired. When the flap control lever is moved, the flap selector valve is positioned by means of cables, pulleys, quadrants and push-pull rods to allow each hydraulic system to supply flow to its associated motor. These motors drive the main flap drive gearbox. This motion is transmitted to torque tubes that rotate screw jacks, mechanically raising or lowering the flaps. The flaps will assume the position selected automatically, without further attendance, by reason of a followup mechanism connected to the selector valve. The selector valve returns to a neutral position when the selected degree of extension or retraction has been reached. Should an asymmetric flap condition occur, a flap phase valve, for each hydraulic system, acts as a shutoff valve which is controlled electrically by action of asymmetric sensing units. The sensing units are installed on the outboard end of each outboard flap screw jack.

Spoiler Hydraulic System

Each inboard spoiler has two hydraulic actuating cylinders and each outboard spoiler has four cylinders. The cylinders are in pairs and are located beneath the spoilers. Each main hydraulic system supplies pressure to one cylinder in each set. The actuating cylinders move the spoilers up or down as required. There is a servo valve assembly for each of the spoilers. Movement of a control lever on the pilots' pedestal operates a cable and pulley system in the aileron-spoiler mixer assembly. The rotational movement of the pulleys is converted to linear motion by push-pull rods and bell cranks. If the air loads reach a critical amount, the spoilers will partially retract, preventing damage to the spoiler surfaces. This spoiler "blow-down" is permitted by the designed internal leakage of the spoiler selector valves. The spoilers are normally operated by both No. 1 and No. 2 hydraulic systems. Should one of the main systems become inoperative, the spoilers may be operated by use of the remaining system. The spoiler operating rate will be approximately the same as normal; however, spoiler "blow-down" speeds will be reduced slightly. Each of the four spoiler surfaces has a servo valve assembly which is used to control hydraulic flow at the actuating cylinders.

Horizontal Stabilizer Trim Hydraulic System

Only the No. 1 main hydraulic system is used to operate the horizontal stabilizer trim system. Movement of one of the trim wheels on the pilots' pedestal rotates a followup shaft which moves a valave to port hydraulic pressure to a hydraulic motor. This motor drive a traveling nut on a screw jack. As this nut moves up or down on the screw jack it causes the leading edge of the horizontal stabilizer to move up or down.

The motor, nut, followup, and valve are connected to the stabilizer. When the stabilizer reaches the position of trim selected, the followup mechanism, riding on the followup shaft, returns the hydraulic valve to a neutral position. An electrically operated valve, controlled by a STAB TRIM HYD SHUTOFF switch in the flight compartment, is incorporated in the hydraulic pressure supply line to the trim motor. Should the trim system become erratic, this valve may be actuated to shut off hydraulic pressure at the trim motor. A standby electric trim motor is available as a backup to the emergency manual system to obtain desired trim settings in the event of a normal system malfunction. This motor, or the emergency manual system, operates the screw in lieu of the nut.

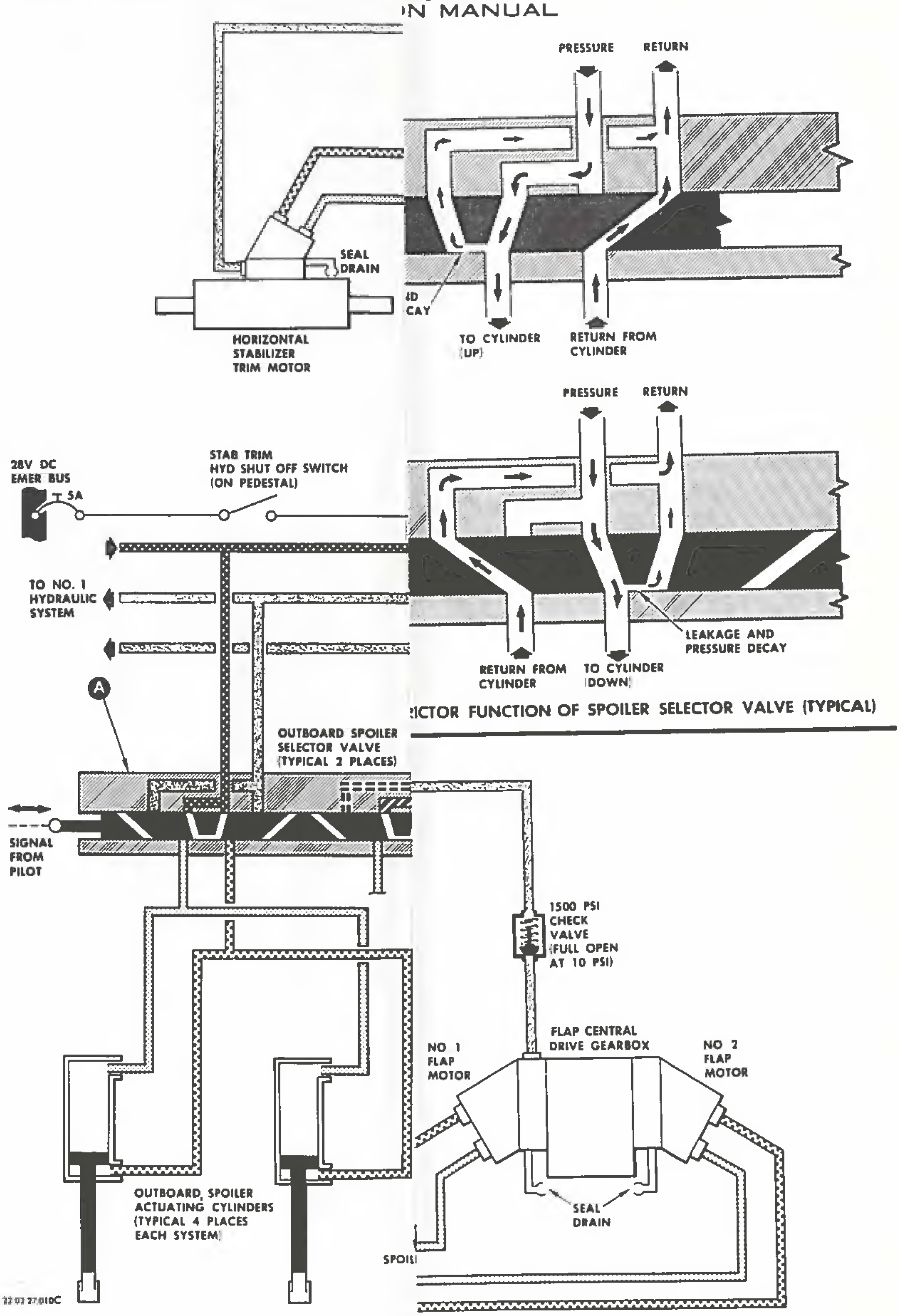
AILERON CONTROL SYSTEM

The aileron system is controlled by movement of the pilot's control wheel (and by the copilot's control wheel through the spring-loaded control column interconnect). Control cables and push-pull tubes mechanically position the flight tabs to initiate aileron movement in the desired direction. A series of push-pull tubes and bell cranks interconnect the ailerons to prevent up-float and down-float. See Figure 7-4 for a diagram of the aileron and spoiler control system. The aileron-spoiler mixer assembly provides a means of coordinating mechanical movement of the ailerons and the spoilers, however, these systems are not interconnected at the mixer assembly. The only interconnection is through the spring-loaded tube assembly connecting the bell cranks at the base of the control columns. A spring-loaded cam and roller unit is incorporated as an integral part of the mixer unit to provide artificial feel for the aileron-spoiler control systems.

Aileron and Aileron Flight Tab Travel

Aileron travel is 15 degrees trailing edge "up" and "down" from the streamline position. The gust dampers also provide up and down stops for the aileron surfaces. When the gust damper is bottomed out against its housing, the aileron is at its full limit of travel.

Flight tab travel is 21 degrees trailing edge "up" and "down" from the streamlined position. Stops are provided to limit tab travel.



Hydraulic System
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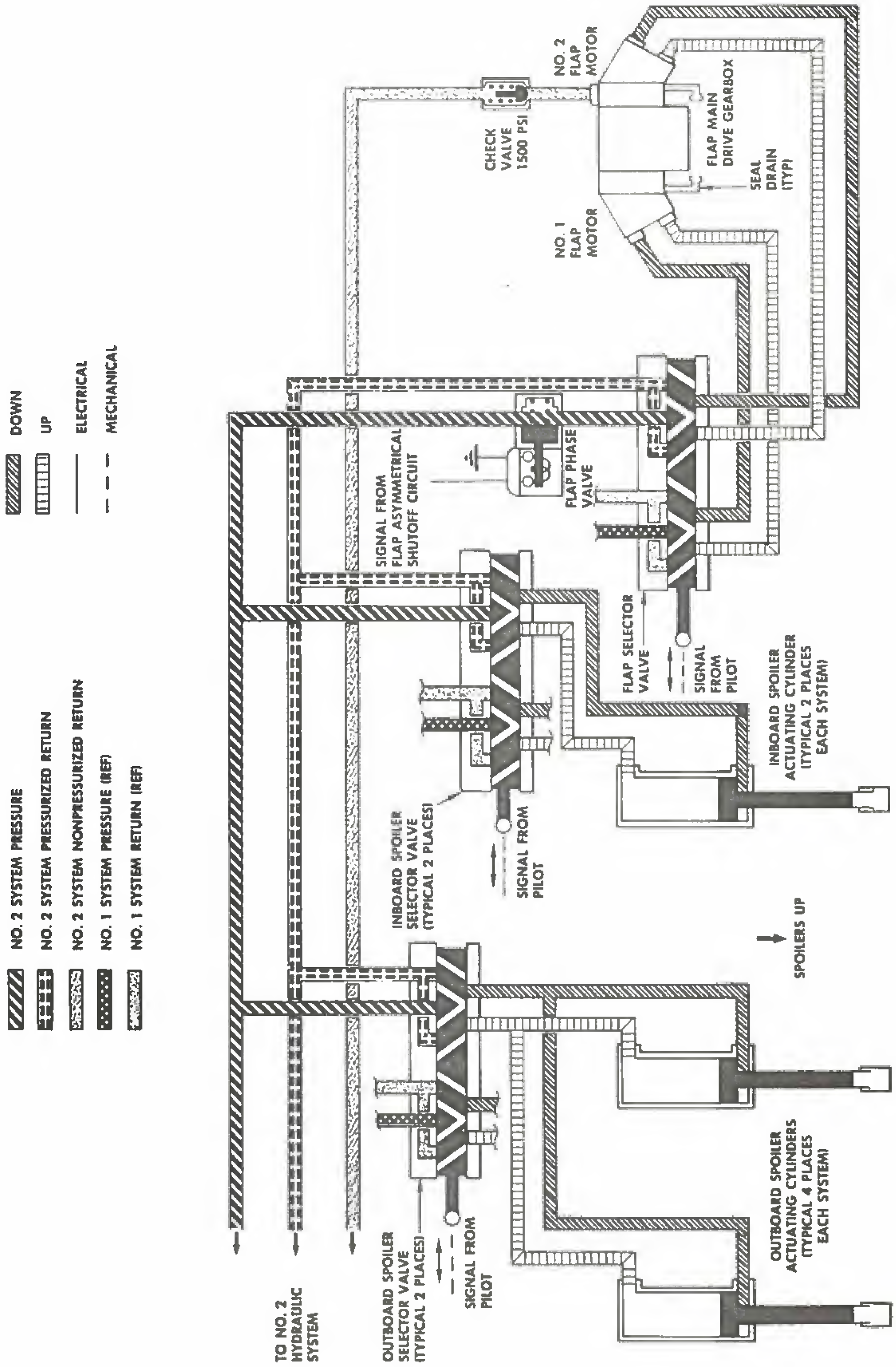
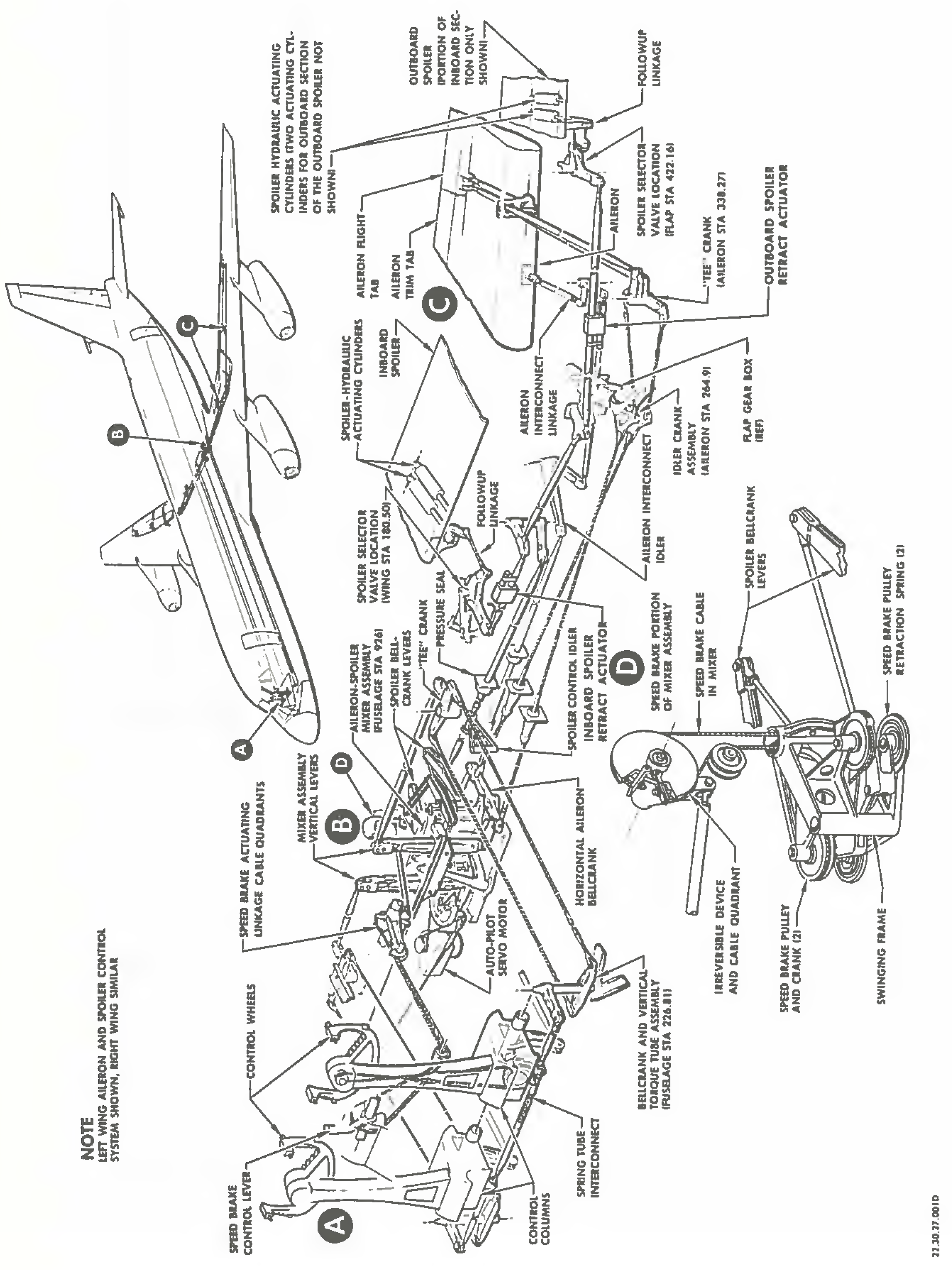
re 7-2



Flight Control Hydraulic System (No. 2 Main System)
Figure 7-3



Aileron and Spoiler Control System
Figure 7-4

Aileron Ground Check System

Since the ailerons are free-floating, with all of the movement of the aileron control wheel being utilized to move the flight tab, an iced or jammed aileron ordinarily could not be detected by the pilot when the airplane is on the ground. A ground check method has been provided by the addition of stops to the front spar of the aileron. The stops are located in such a manner that after 60 degrees of aileron wheel movement, the bell crank located on the aileron hinge line, when moving for down aileron, contacts a checkout stop and as a result, moves the aileron to its full travel. Since the ailerons are interconnected, the opposite aileron will also move. However, in flight the flight tabs will cause the aileron to move and the checkout stops on the aileron will move away from the bell cranks on the aileron hinge line.

Aileron Balance Boards

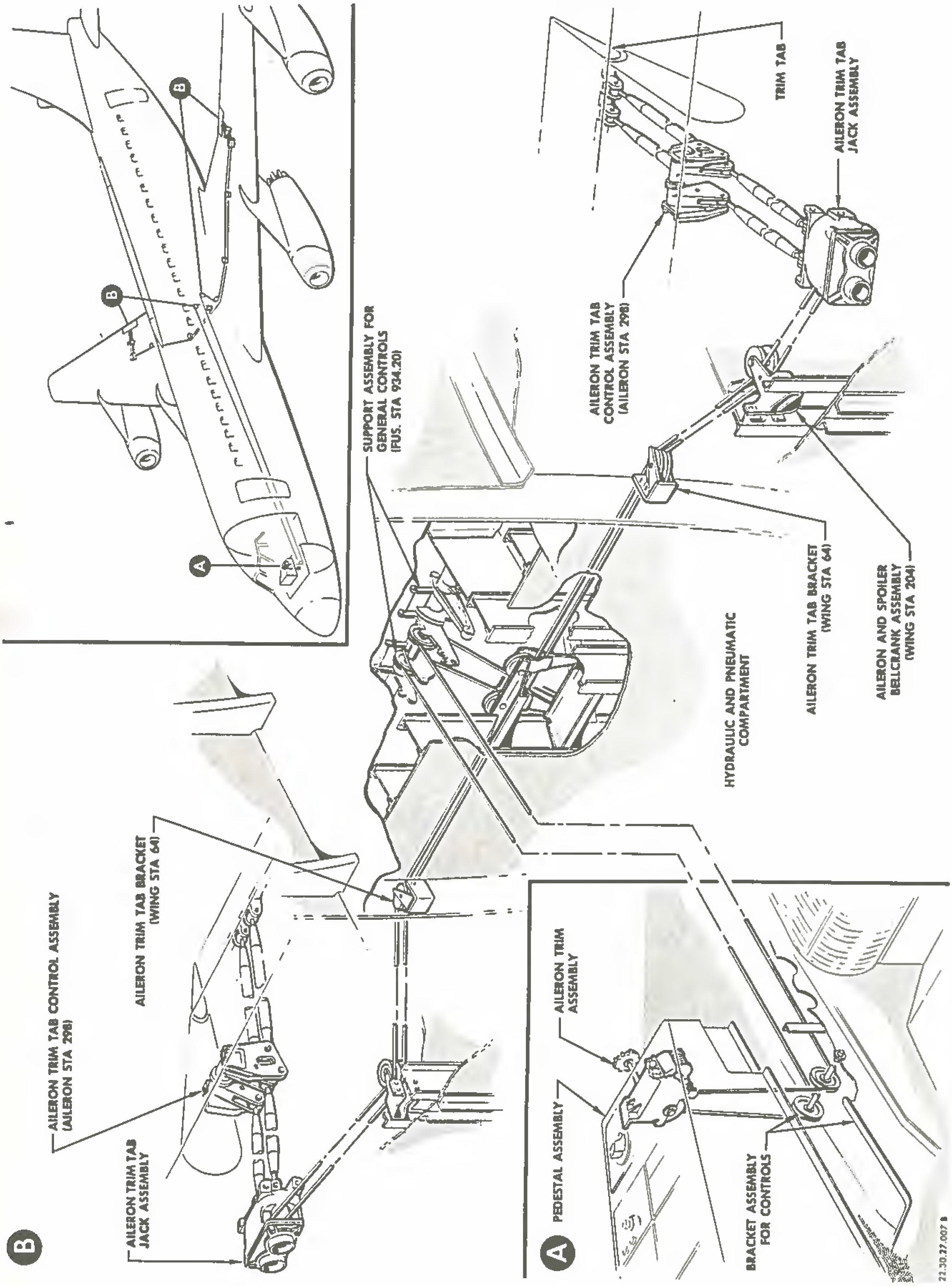
Balance boards and balance weights are attached to the leading edge of the ailerons to give the ailerons aerodynamic and dynamic balance. Curtains are installed between the balance boards and the wing trailing edge structure to provide seals for the balance chambers.

Aileron Trim System

The aileron trim tabs are controlled by a trim knob located on the aft side of the pilots' pedestal (see Figure 7-5). Movement of the aileron trim knob also positions the aileron trim indicator scale on the pilots' pedestal to provide an indication of the amount of the trim introduced. Aileron trim tab travel is 16 degrees "up" and "down" from the streamline position to 7.5 degrees with ailerons fully extended. Stops limit trim tab travel.

SPOILER CONTROL SYSTEM

Two spoilers, mounted on the upper surface of each wing, operate in conjunction with the ailerons and provide the principle means of lateral control. The spoilers are also extended symmetrically to serve a secondary function as speed brakes. Each inboard spoiler is composed of two interconnected sections which move as a single surface. The inboard spoiler is actuated by a hydraulic actuator at each section and is controlled by a dual hydraulic selector valve. Each section of the outboard spoiler, composed of two sections, utilizes four hydraulic actuators. The outboard spoiler actuators are also controlled by a dual hydraulic selector valve. Selector valve operation is provided through a push-pull tube system connected to each side of the aileron-spoiler mixer assembly. An additional push-pull rod is attached between each spoiler, and a



Aileron Trim System
Figure 7-5

followup linkage completes the loop for closing the selector valves when the spoilers reach the selected degree of travel. As a safety feature, power for operating the spoilers is obtained from both main hydraulic power systems. The spoiler selector valves incorporate a "blow down" feature which provides structural protection of the spoilers. However, for normal operation the speed brake handle should not be moved aft of the placarded normal operating range. For emergency longitudinal trim (jammed horizontal stabilizer) the speed brake handle can be moved aft into the placarded restricted area (crosshatch) but not aft of the double white lines. In this event, with inboard spoilers extended the speed restrictions as published in the FAA approved Flight Manual must be observed. For emergency descent and landing roll braking the speedbrake handle can be moved beyond the double lines to the full aft position for full spoiler extension.

When the speed of the airplane exceeds approximately 220 knots IAS, "blow down" is permitted by the spoiler selector valves, thereby preventing structural damage due to excessive airloads. Should one of the hydraulic systems lose pressure, hinge moment will be reduced and "blow down" will occur at lower speeds; but full deflection will still be possible at approximately 150 knots IAS or below.

The spoiler and aileron systems are not interconnected at the mixer unit. The only interconnection is through the spring-loaded tube assembly connecting the bell cranks at the base of the control columns. Force applied at the left-hand control column wheel is transmitted to the aileron tabs through the linkage on the left-hand side of the airplane. The force is also carried through the spring tube assembly into the linkage on the right-hand side of the airplane to the mixer unit and finally to the spoiler control valves. The force at either aileron control wheel causes the spoiler surfaces to move.

Separate linkages for the spoiler and aileron flight tabs provide a safety feature. During normal operating conditions, either control wheel operates both systems. Should a malfunction occur causing one system to jam, the system on the opposite side can be operated by overriding the jam through the spring-loaded tube assembly. A jam in the spoiler system on the right-hand side of the airplane can be overridden by the pilot to operate the aileron system. A jam in the aileron system can be overridden by the copilot to operate the spoiler system.

Aileron-spoiler relationship is as follows: With the spoilers flush against the upper wing surface (down position), only the spoilers on the up-aileron side will move. With the spoilers full open (up position), only the spoilers on the down-aileron side will move. With the spoilers selected to speedbrakes to any position other than full up or full down, the spoiler direction of movement will be the same as with both ailerons; that is, one set will move up farther and the other set will move down farther.

Separate Spoiler Control

The inboard or outboard spoilers can be retracted separately at the option of the pilot. This is accomplished by electrical actuators located in the valve input rod linkage. A switch on the pilots' pedestal enables the pilot to select the desired pitch-up or pitch-down mode, controlled in magnitude by the

speedbrake control lever. The function of the system is to provide longitudinal trim by means of the speedbrake control lever, in the event of a horizontal stabilizer trim system failure. (See Section 2, EMERGENCY PROCEDURES, of the Airplane Flight Manual.)

Spoiler Travel

Maximum spoiler travel is 60 degrees "up" from the wing upper surface. Stops are provided in the wing trailing edge structure to limit the retracted travel position of the spoilers.

ELEVATOR CONTROL SYSTEM

The elevators are used for longitudinal control of the airplane while in flight and are controlled by a mechanical system of push-pull rods, bell cranks, and cables that connect the control columns in the flight compartment to the flight tabs on the elevators (see Figure 7-6). The right and left-hand elevator control systems are interconnected by two spring tube assemblies. One spring tube assembly is located between the first two bell cranks aft of the control columns; the other spring tube assembly is located between bell cranks in the aft section of the elevator controls. In the event of a jam in one side of the elevator control system, the use of spring tube interconnects allows the pilot on the opposite side of the jam condition to maintain partial elevator control.

Elevator and Elevator Tab Travel

Elevator travel is 25 degrees "up" and 12 degrees "down" from the streamline position. The elevator gust damper also acts as a limit stop for the elevator. Flight tab travel is 13 degrees "up" and 26 degrees "down" from the streamline position. Travel is limited by flight tab stops.

Elevator Balance Boards

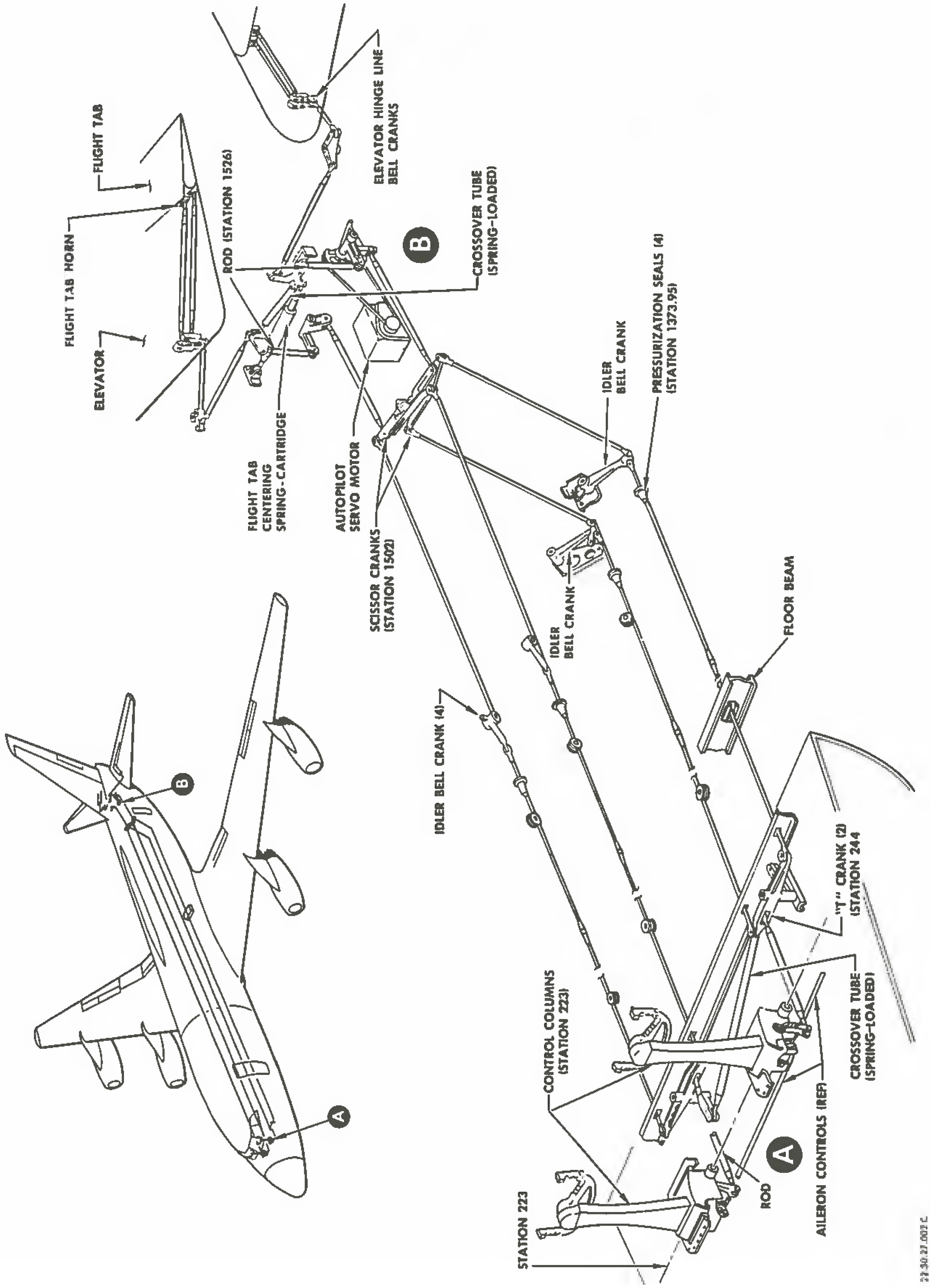
Through a system of parallelogram linkage, three balance boards (see Figure 7-7) are connected to the front spar of each elevator. On movement of the elevator a pressure differential occurs in the sealed balance board chambers through orifices along the stabilizer trailing edge ahead of the elevator hinge. Three sets of parallelogram linkages are installed between the elevator and the balance boards at the hinge attachment fittings of each elevator.

Elevator Flutter Dampers

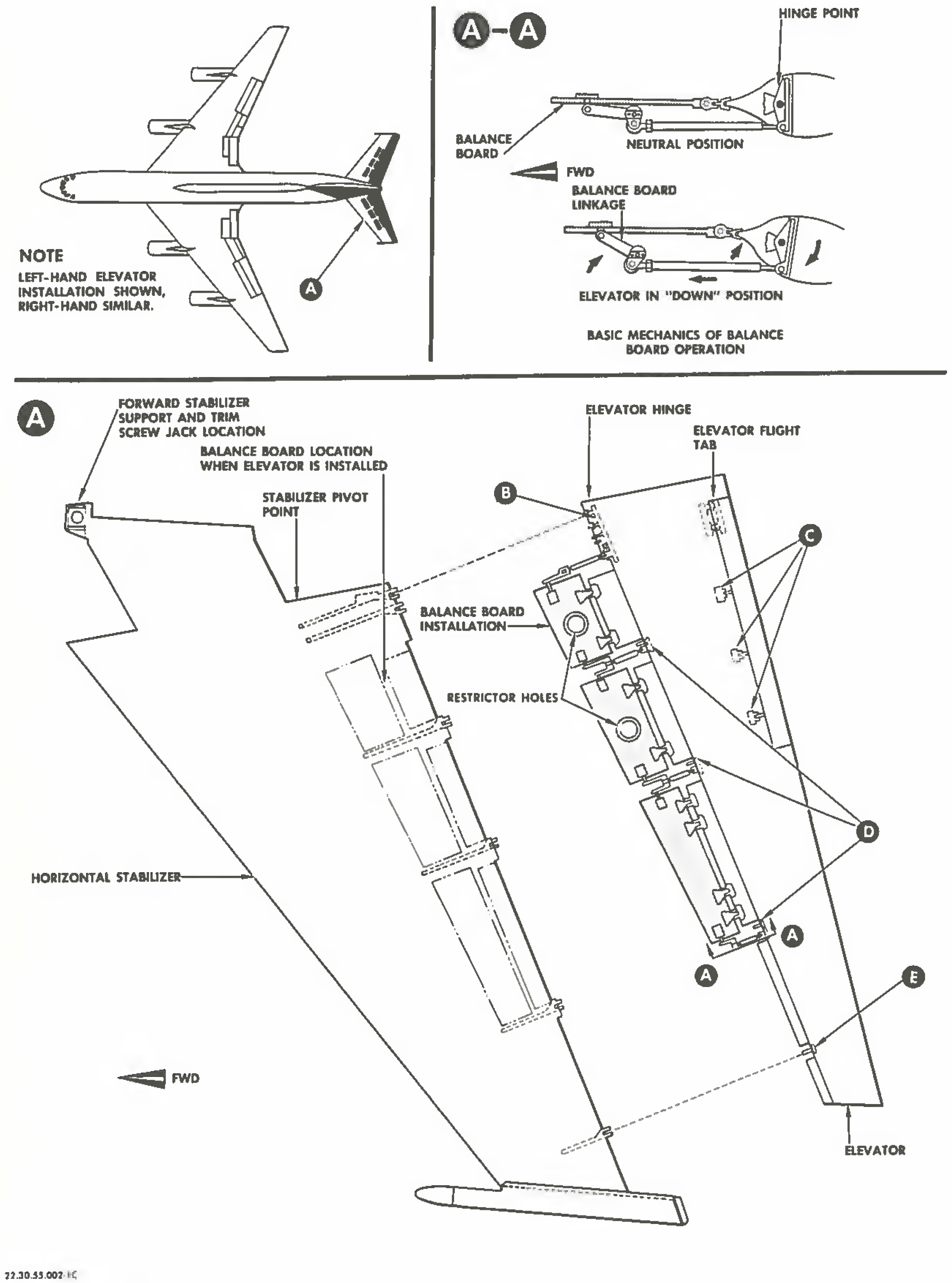
The flight tabs have hydraulic type flutter dampers attached to the linkage near each tab. The dampers eliminate any tab flutter which might occur in flight. The elevator surfaces have hydraulic type flutter dampers.

HORIZONTAL STABILIZER TRIM SYSTEM

In order to avoid the problems encountered in using an elevator trim tab at high airplane speeds, a completely movable horizontal stabilizer is used. Moving the entire stabilizer also provides the large trim moments necessary at take off and landing, and the design is aerodynamically clean at cruise speed. The normal mode of horizontal stabilizer operation consists of three subsystems:



Elevator Control System
Figure 7-6



Elevator Installation
Figure 7-7

Electric trim control, Manual trim control and Autopilot trim control. The emergency mode of operation is divided into two subsystems: Electric trim control and Manual trim control. In addition to the normal and emergency modes of operation, a Speed Stability system is used. This system functions only when the airplane is flying at speeds above 200 knots with an additional input in the speed range between Mach 0.80 through Mach 0.91 and is completely automatic. An irreversible Acme type screw jack assembly regulates the positioning of the stabilizer during all conditions of normal, emergency and speed stability system operations.

Horizontal Stabilizer Travel

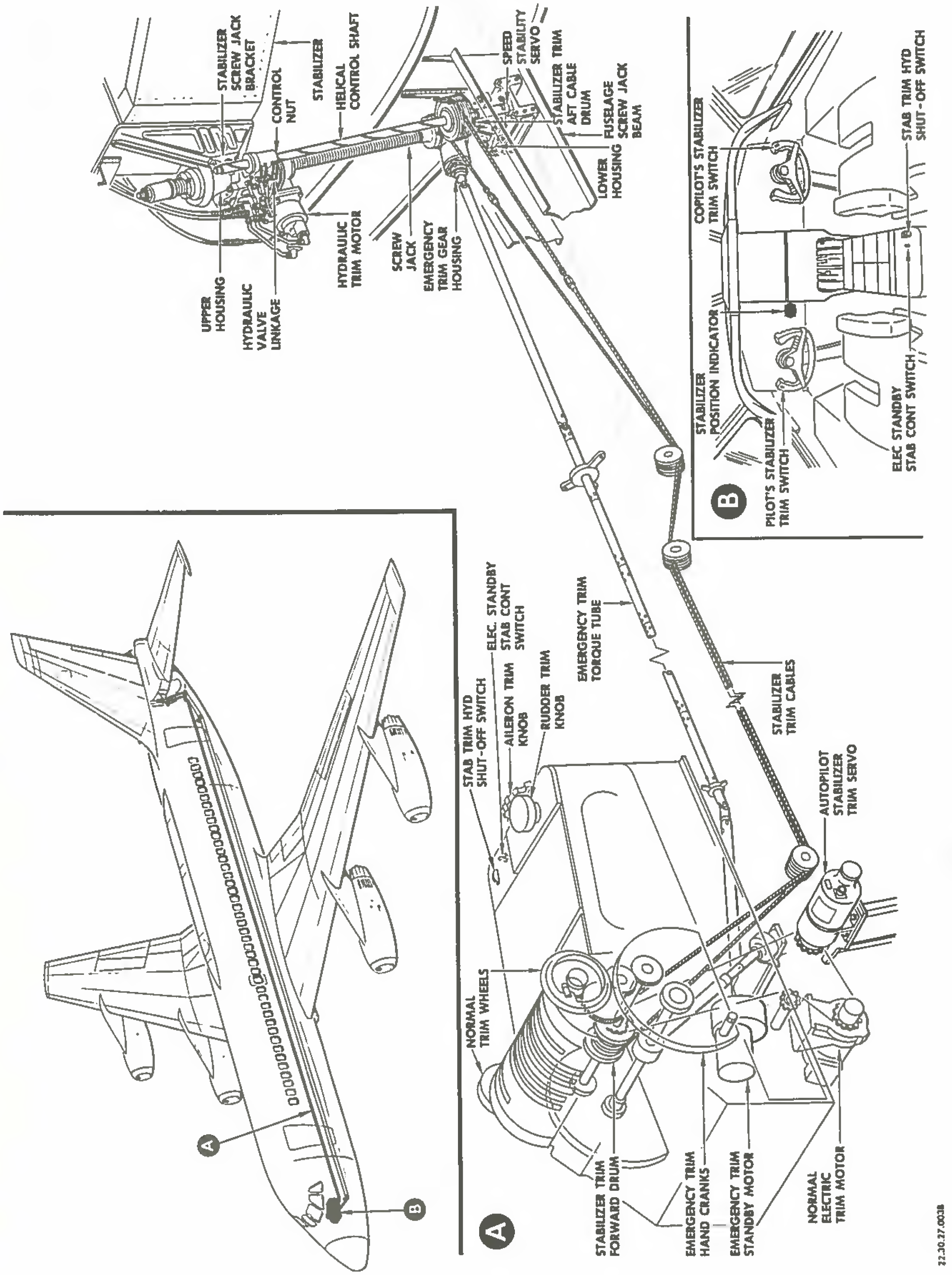
The amount of trim permissible in using the horizontal stabilizer is limited by safety requirements. Maximum horizontal stabilizer deflection is from streamline to 14 degrees stabilizer leading edge down (resulting in a maximum 14 degree airplane nose up trim. Stabilizer rate is automatically controlled by dynamic pressure. At airplane speeds above 250 knots, the rate is reduced to give the pilot sufficient time to take safety measures if a failure should occur. The stabilizer movement rate is 0.4 degree per second up to 250 knots, reduced automatically to 0.2 degrees per second at higher speeds, and 0.1 degree per second when on autopilot.

Normal Stabilizer Trim Control Systems

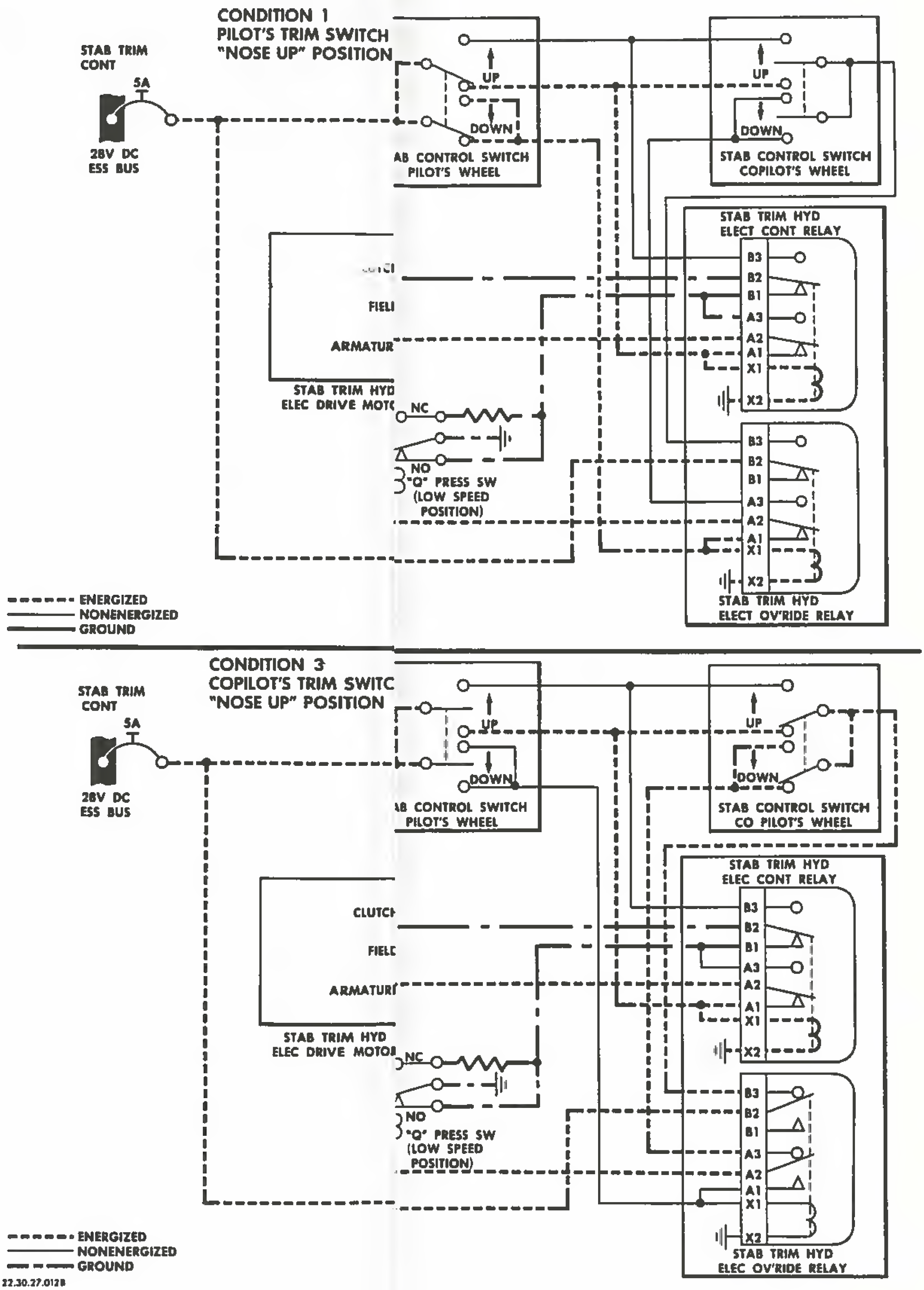
Normal stabilizer trim control is accomplished manually by movement of the upper trim wheels (see Figure 7-8), located on the forward right and left sides of the pilots' pedestal or electrically by actuating a sliding switch on the pilot's or copilot's control wheel (see Figure 7-9). During normal operation, manual movement of the trim wheels or electric actuation of a trim motor moves a cable drum to which the control wheels and trim motor are geared. The movement of this forward cable drum actuates a cable system that is then routed to a cable drum on the screwjack assembly. Movement of the cable drum on the screw jack actuator is the input force which actuates the hydraulic selector valve. During normal operation, a traveling nut attached to the front spar of the stabilizer rotates around the screw jack to provide stabilizer trim movement. The traveling nut, by means of a worm gear, is rotated by a hydraulic motor. Movement of the trim wheels (manually or electrically) also positions the dual trim indicator drum dials on the pilots' pedestal.

Control Wheel Switch Trim System

An electrically operated motor and clutch under the pilots' pedestal is connected into the normal operating cable system at the cable drum by means of the same chain and torque tube drive as the autopilot trim motor. Control is provided by means of sliding contact type switches installed on a knob on the upper part of the outboard horn of the pilot's and copilot's control wheels. The switches operate up and forward for nose down trim and down and aft for nose up trim. This trim system can be disarmed by actuation of a circuit breaker in the flight compartment. The control wheel trim system is inoperative when operating on autopilot.



Horizontal Stabilizer Trim Control System
Figure 7-8



ilizer Trim Control System
gure 7-9

Stabilizer Emergency Trim Control Systems

The standby electric trim motor or the emergency trim control wheels can be used by the pilots to obtain the desired trim setting in the event of a malfunction of the normal trim systems.

For emergency operation, the traveling nut on the screw jack assembly remains stationary and the screw jack is rotated. This is accomplished by turning either or both of the two large emergency trim control wheels mounted on each side of the pilots' pedestal below the normal trim control wheels. Through a system of gears and torque tubes, emergency trim wheel forces are routed aft to the gear box at the base of the screw jack assembly. The lower gear drive is independent of the hydraulically operated upper gear drive used for normal trim control. In order to operate the emergency stabilizer actuator by manual control, the hydraulic power to the actuator must be shut off. This is accomplished by actuating the STAB TRIM HYD SHUT OFF switch, located on the pilots' pedestal, to the CLOSED position.

A reversible ac electric servo motor gear drive is installed beneath the pilots' pedestal on the forward end of the manual emergency system torque tube. This motor can be used to actuate the emergency trim system instead of manual operation of the emergency trim wheels. Maximum trim rate is approximately 0.1 degree per second. Two switches are required for such operation. The STAB TRIM HYD SHUT OFF switch must be placed in the CLOSED position. This shuts off the hydraulic power to the stabilizer mechanism and arms the ELEC STANDBY STAB CONTROL switch. This switch is held in the NOSE DN or NOSE UP position until the desired degree of trim has been obtained.

Horizontal Stabilizer Trim Indicators

A drum dial trim indicator is located inboard of each normal stabilizer trim control wheel on the pilots' pedestal. These indicators are connected to the cable system and are calibrated to read in degrees of stabilizer movement. In addition to the drum type indicators, a stabilizer position indicator dial is located on the pilot's instrument panel. This indicator is a synchro motor operated by a synchro transmitter located on the right drag longeron in the tailcone of the airplane. An arm on the input shaft of the transmitter is connected to one end of a push rod; the other end of the push rod is connected to the rear spar of the horizontal stabilizer. The instrument panel indicator indicates the true position of the stabilizer. The drum dial indicators on the pedestal indicate only the cable travel. Cable travel is directly proportional to the stabilizer movement except for the limited input of the speed stability augmentation system. This input causes a maximum dial indicator error of approximately 3.7 degrees.

Automatic Pilot Horizontal Stabilizer Trim System

When it is turned on, the automatic pilot operates a 115/200 volt reversible ac motor drive gear located beneath the pilots' pedestal. The motor drive gear is connected to the normal control operating cable drum by a chain drive and a

torque tube. When the autopilot system is turned off, the motor gear drive is electrically declutched so the actuator operation can be accomplished by normal operation control of the cable system to the hydraulic motor. The maximum rate of trim provided by this motor is approximately 0.1 degree per second.

Speed Stability Augmentation System

Automatic stability of the airplane is maintained at airspeeds between Mach 0.80 and 0.91 and above 200 knots EAS by the speed stability augmentation system. High speed airplanes of the Convair 880 type, when flying at airspeeds between these Mach numbers, have a reverse aerodynamic tendency commonly known as "tuck under effect". The tendency is to nose down when speed is increased. In addition to the pilots' normal mode of controlling the horizontal stabilizers' angle of attack, the horizontal stabilizer screw jack assembly incorporates a reversible servo motor tachometer generator and a feedback potentiometer. These components are enclosed in a housing attached to the screw jack assembly and are geared into a differential mechanism to perform their particular functions in the speed stability augmentation system. In addition to these components, the speed stability augmentation system has a servo power amplifier, monitor and a flight air data computer.

The energized flight air data computer senses pitot pressure, static pressure, outside air temperature, and altitude. It combines these data to compute a true Mach number and dynamic pressure. The computed power signal positions a wiper on a potentiometer in the flight air data computer. The wiper position, which is determined by the magnitude of the computed power signal, commands respective servo motor movement in the Mach 0.80 to 0.91 range and for EAS speeds above 200 knots. In this range, the energized amplifier senses and amplifies any difference in excitation voltages caused by the difference between positions of the air data computer wiper and the feedback potentiometer wiper, thereby driving the electric servo motor. The electric servo motor stops when the horizontal stabilizer reaches the position of trim commanded by the air data computer.

The differential drive is a part of the stabilizer screw jack and contains a built in frictional drag to eliminate back lash in the gearing for either pilot or speed stability augmentation inputs. This differential drive also contains a mechanical stop which limits stabilizer travel to 3.7 degrees to prevent exceeding a controllable trim change of the airplane due to failure of the speed stability system. The electric servo motor drives the followup shaft to actuate the hydraulic valve. The hydraulic valve ports hydraulic pressure to the hydraulic servo motor to rotate the traveling nut on the screw jack. This movement positions the horizontal stabilizer to the trim position commanded by the air data computer and "Q" transducer.

A speed stability fail indicator light is provided on the pilot's instrument panel and is connected into the MASTER CAUTION light system.

Horizontal Stabilizer Position Warning Horn

The stabilizer position transmitter incorporates a cam on the input arm which closes a switch when the stabilizer is between 0 and $1\frac{1}{2}$ degrees and between 7 and 12 degrees airplane nose up position. This switch is in series with a switch on the throttles and a switch on each main landing gear strut (see Figure 7-10). A warning horn in the flight compartment will sound intermittently if the stabilizer is not in the take off range and any two power levers are advanced past the 75 percent RPM position with the main landing gear struts compressed.

RUDDER CONTROL SYSTEM

The rudder is hinge mounted to the rear spar of the vertical stabilizer and is controlled by movement of the rudder flight tab through the pilot's or copilot's rudder control pedals. This action positions the flight tab at an angle to the rudder centerline opposite to the desired deflection and initiates rudder surface movement in the desired direction.

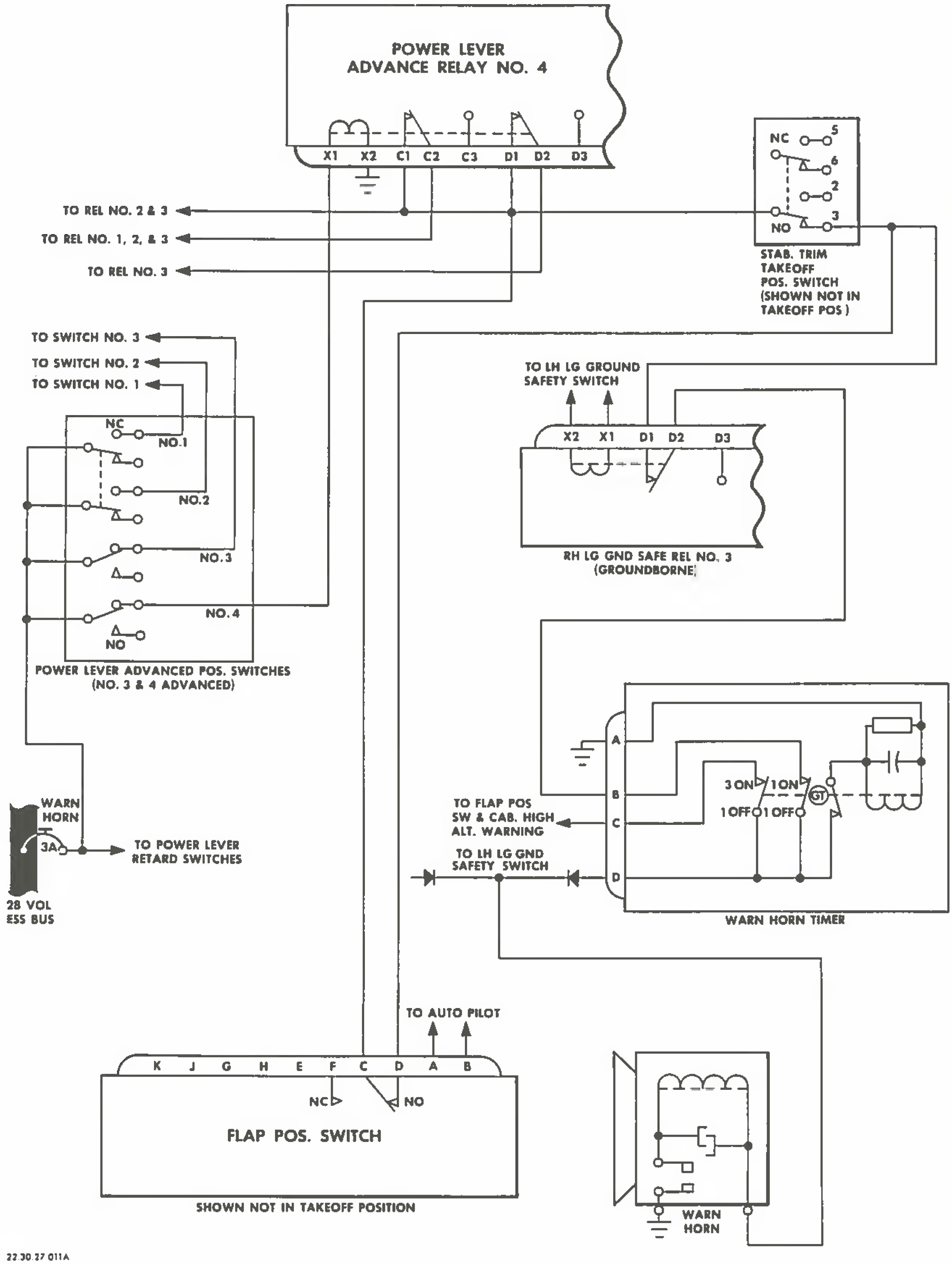
To prevent the pilot from inadvertently deflecting the rudder excessively at high airplane speeds, a pedal force limiter is incorporated in the rudder control system. A preloaded spring in the limiter begins to progressively collapse when the pressure on the rudder pedals exceeds 180 pounds. The rudder controls are rigged so that at 225 knots IAS, 180 pounds pressure on the rudder pedal will provide full flight tab deflection. When pedal pressure exceeds this amount, the spring compresses at a rate of approximately one inch per each additional 45 pounds pressure as measured at the rudder pedals. The rudder pedals, contacting their stops will therefore limit full rudder deflection to airplane speeds of not more than 225 knots IAS.

Rudder Slip Sensing Device

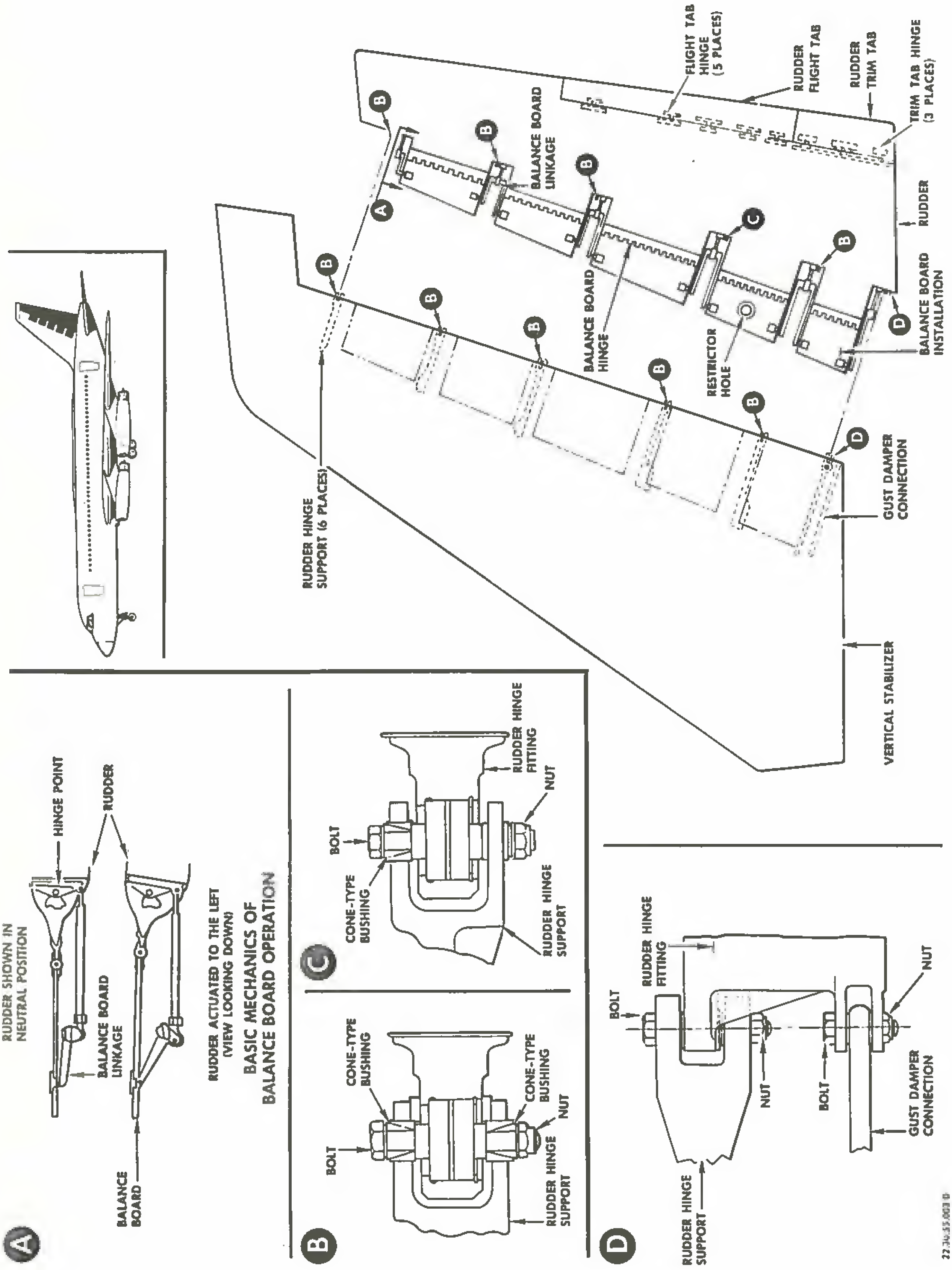
A side slip sensing pressure box is incorporated in the rudder system. The pressure box is connected in the system in such manner that a side slip condition is sensed by the box which exerts a force in the control linkage to command a slight tab movement to correct the condition. Force for the operation of the pressure box is supplied by the differential pressure across the vertical stabilizer, and is a function of airspeed, altitude and angle of sideslip. The sideslip pressure box, in conjunction with a pre-loaded feel spring, provides satisfactory feel forces to the pilot when he is applying rudder control, whether in zero sideslip or sideslip conditions. A "Beta Box" test switch and indicator light is provided on the pilot's auxiliary instrument panel. The test switch is held ON while imparting a rapid input to the rudder panels. The green light will illuminate if the Beta Box is operating properly.

Rudder Balance Boards

A series of five balance boards (see Figure 7-11) are attached to the rudder along its vertical leading edge. These boards move in compartments in the trailing edge of the vertical stabilizer. The compartments are sealed except at the aft end, where there is a 0.75-inch gap between the rudder and the stabilizer shroud doors.



Stabilizer Warning Horn Schematic
Figure 7-10



Rudder Installation
Figure 7-11

The boards are attached to the rudder by a parallelogram type mechanism which allows the balance boards to move to the right or left. On movement of the rudder, a pressure differential occurs in the balance board chambers, thus exciting a boost force to help in deflecting the rudder.

Rudder and Rudder Flight Tab Travel

Rudder travel is 18.5 degrees to the left or right along the vertical stabilizer chordline. The travel is limited by bottoming of the gust damper. Flight tab travel, along the rudder chordline, is 21 degrees to the left or right. Travel is limited by flight tab stops.

Rudder Pedal Travel

Rudder pedal travel, due to the maximum flight tab travel, is 1-3/8 inches forward and aft from neutral; pedal travel, due to combined rudder and flight tab travel, is 3 1/3 inches forward and aft of neutral. The pedals are adjustable through a 7 inch range. Stops are provided on pedals and trim cables to limit travels. Individual rudder pedal adjustment is provided by a hand crank located between and aft of each pair of rudder pedals. Adjustment of one set of rudder pedals has no effect on the other set, nor any effect on the operation of the rudder system.

Rudder Flutter Dampers

Dual hydraulic flutter dampers are provided for the rudder flight tab. Three hydraulic flutter dampers and a gust damper are provided for the rudder.

Autopilot Rudder Operation

In addition to the pilot input during manual flight operations, the rudder flight tab receives input from the autopilot servo motor, which is controlled by the yaw damper and turn coordinator system. Yaw rate is sensed by the primary autopilot control. The signal from the sensors directs a rudder movement proportional to the yaw disturbance. The yaw damper-turn coordinator and the autopilot are controlled by a three-position switch in the flight compartment. During autopilot operation, the autopilot servo motor input is transmitted to the autopilot quadrant, located in the vertical stabilizer, by means of a cable-loop system.

FLAP CONTROL SYSTEM

Two double-slotted, Fowler-type flaps are installed in the trailing edge of each wing, one inboard and one outboard of the aileron. The flaps are so designed as to provide low-drag/high-lift when partially extended, and high-lift/high-drag when fully extended. The fully extended position of the flaps is 50 degrees. (For maximum airspeeds and flap operation, consult the Airplane Flight Manual.)

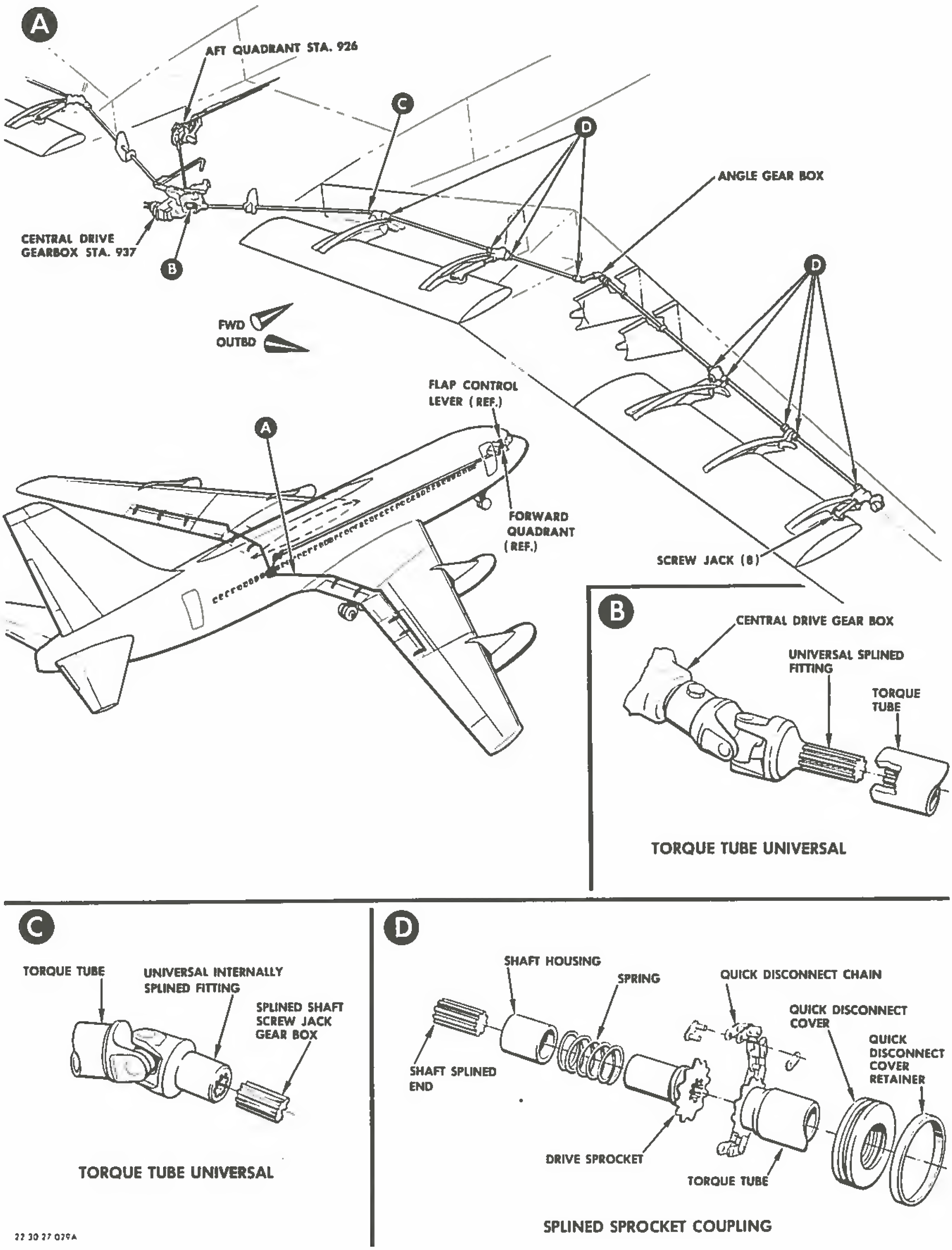
The flap extending mechanisms are contained within the wing trailing edge overhang, providing a minimum of exposed units to create parasitic drag. The flaps are attached, through roller equipped carriages, to curved flap support tracks. The outboard flap support tracks are attached to the wing rear spar. The inboard flap support tracks are attached to the main landing gear beam in the wing. The flaps are moved on these tracks by recirculating ball bearing screw jacks operated through gearboxes driven by a torque tube drive system. Extension of the flaps can be controlled to geometrically widen the chord through the first part of flap extension and to increase the drag coefficient rapidly during the last part of flap extension. The Fowler flap incorporates a double slot in its leading edge which controls airflow over the flap to provide maximum effectiveness when it is extended into its landing position. The flap extension screw jacks tie into both the inboard and outboard ends of each flap and extend or retract the flaps according to the direction of screw jack rotation. Power to rotate the torque tubes is furnished by two hydraulic motors mounted on a main drive gearbox. A dual hydraulic selector valve is mechanically controlled by a lever on the right side of the pilots' pedestal. This lever is connected to the dual hydraulic servo valve by a cable loop and push-pull rod system. This mechanism regulates hydraulic motor power to the central drive gearbox (see Figure 7-12).

Flap Main Drive Gearbox and Motors

The main drive gearbox is located in the hydraulic compartment at the airplane centerline. Two hydraulic motors are utilized. These two motors are of the constant displacement, piston type and are mounted one to each side of the gearbox. Hydraulic power to the two drive motors is supplied by both main hydraulic power systems; one system furnishing fluid to one motor, the other hydraulic system furnishing fluid to the other motor. This provides a safety back-up feature in the event that one hydraulic system should fail.

Flap Operation Speeds

The input speed of the hydraulic drive motors during the flap extension cycle is approximately 4900 revolutions per minute. The speed of the main drive gearbox output shaft, which drives the torque system, is 1056 rpm during the extension cycle. This output shaft, during the extension cycle, rotates 220 revolutions in order to move the flaps from full retraction to full extension. The time required for the extension cycle is approximately 12.5 seconds. During the retraction cycle, the specific values stated for the extension cycle are reduced by 50 percent. This reduction is accomplished by the internal porting arrangement of the flap servo control valve, which in effect, is only allowing half as much fluid flow to the hydraulic valve motors. The time required for the flaps to travel from full extension to full retraction is approximately 25 seconds.



Wing Flap System
Figure 7-12

Flap Asymmetrical Shutoff Switch

An asymmetric unit is installed at the outboard end of the left and right wing flap torque tube drives. An asymmetrical shutoff switch is a part of each of these units. The units are electrically interconnected, and connected to, the flap phase valves through the flap asymmetrical control relay (see Figure 7-13). Each asymmetrical switch has three cams which are geared to the flap torque tube drive. Each cam has three lobes, the high point of each lobe is 120 degrees apart. Each cam moves a roller actuator to make contact with a stationary contact point. Under normal conditions, the flaps are in symmetry and there is no electric power at the flap phase valves. Any condition that causes the flaps to be out of symmetry by 2 degrees will cause one cam of each asymmetrical switch to move its roller actuator. When contacts are made simultaneously at the proper points, a circuit is completed from the 28-volt dc emergency bus through the two asymmetrical switches and the control relay to the solenoids of the flap phase valves. The solenoids then close the valves, shutting off hydraulic pressure to the flap motors on the main drive gearbox. Flap movement is thus stopped immediately.

Flap Position Indicators

A dual-reading flap position indicator is installed on the center instrument panel in the flight compartment. Two position transmitters are provided, one on each outboard end of the left and right outboard flap. Two indicating needles are used on the indicator: one red needle for the right-hand flap and one white needle for the left-hand flap. The red needle is directly behind the white needle. When the flaps are moving in unison, the red needle stays hidden behind the white needle. However, if a split-flap (asymmetrical) condition develops, the two needles separate and the white needle will uncover the red needle. The dial is calibrated in ten degree increments.

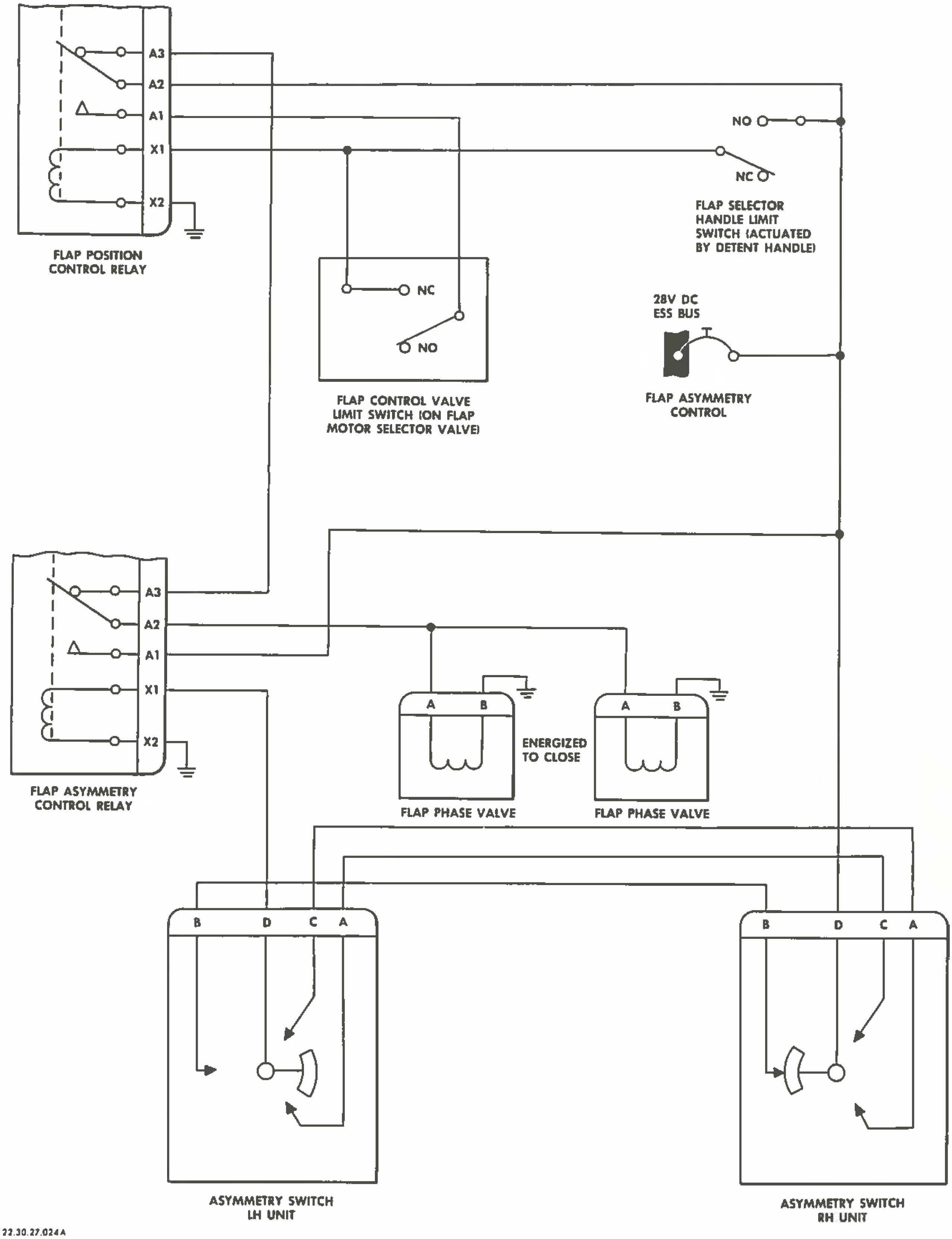
A flap position scale marked off in ten degree increments is located adjacent to the flap control lever on the pilots' pedestal.

Flap Warning Horn Circuit

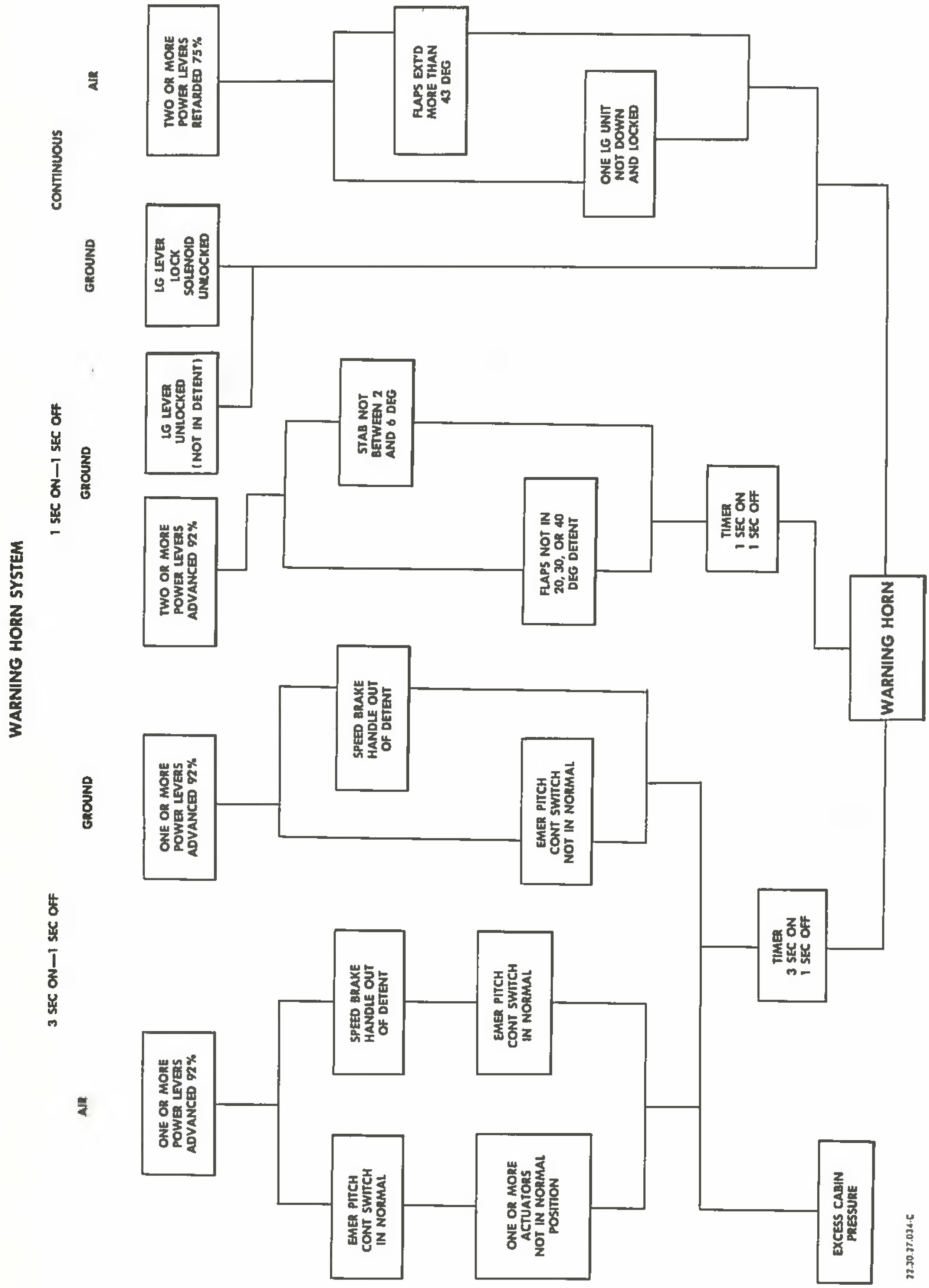
An electrical warning horn circuit is interconnected between the flap position switches, the landing gear safety switches, and the four engine power levers (see Figure 7-14). The warning horn sounds intermittently when the flaps are not in the 20°, 30° or 40° takeoff position and any two power levers are advanced beyond 92 percent power settings. During approach conditions, if any landing gear is not down and locked, lowering the wing flaps beyond 32 degrees of extension will produce a steady horn blast.

CONTROL SURFACE GUST DAMPERS

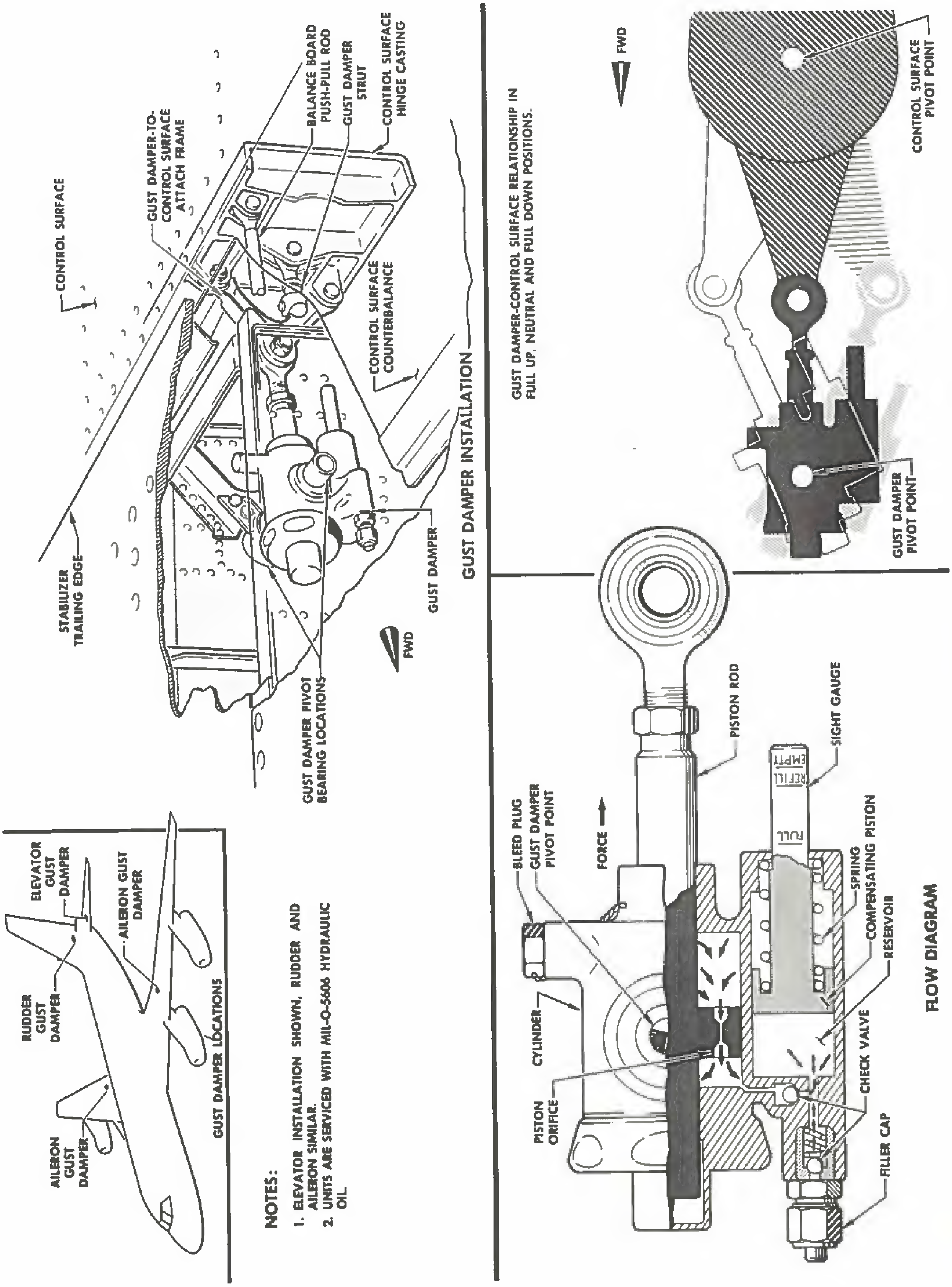
A hydraulic gust damper system provides control surface gust protection when the airplane is on the ground. Each of the three primary control surfaces use the hydraulic gust dampers. The dampers are self contained units and are not connected to the airplane hydraulic system (see Figure 7-15). The dampers are



Flap Asymmetrical Shutoff Switch and Phase Valve Circuit
Figure 7-13



Flap Warning Horn Schematic
Figure 7-14



Control Surface Gust Damper
Figure 7-15

permanently installed and are designed so that damping rates do not effect normal flight movements of the control surfaces. The rudder damper also includes a snubber which dissipates any remaining rudder travel after the initial damping action.

Gust Damper Locations

There are five gust dampers installed on the airplane as follows:

1. Two elevator gust dampers are located at the inboard trailing edge of the horizontal stabilizer at BL 39.37.

2. Two aileron gust dampers are located in the wing trailing edge just inboard of the aileron hinge at station 259.3.

3. One rudder gust damper is located at the base of the vertical stabilizer trailing edge at approximately WL 164.

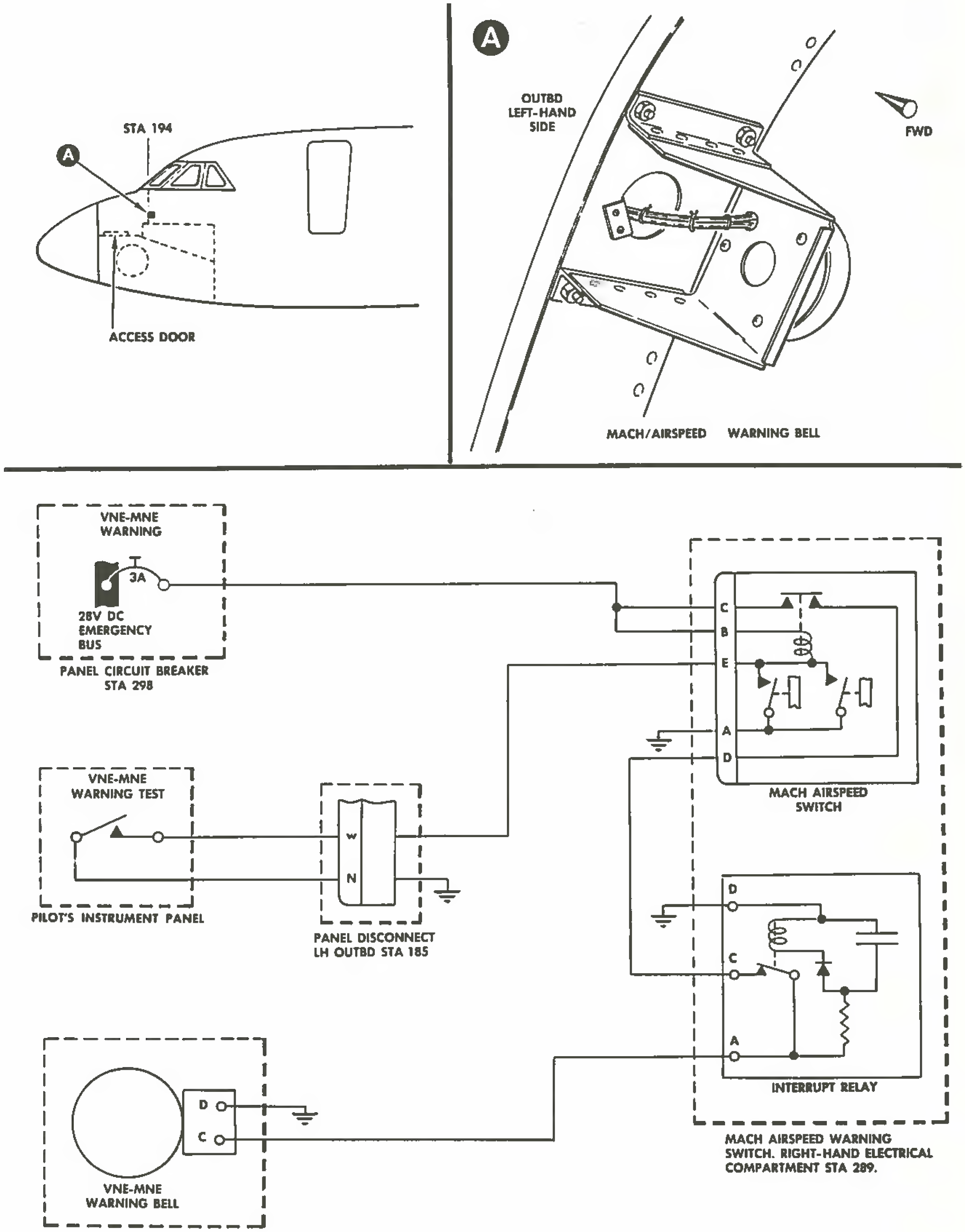
MACH/AIRSPEED NEVER EXCEED WARNING SYSTEM

When the airplane is approaching the maximum allowable Mach or maximum allowable indicated airspeed, a warning bell (to the left and forward of the pilot's seat) rings intermittently. This system is commonly referred to as V_{NE}-M_{NE} for velocity never exceed and Mach never exceed. The system consists of an aneroid Mach/airspeed switch, an interrupter relay, a test switch, and the warning bell (see Figure 7-16). When near maximum Mach or near maximum indicated airspeed is reached, a pressure switch in the lower right-hand electrical compartment is actuated to complete a circuit from the 28-volt dc emergency bus to the relay coil in the pressure switch. The energized relay closes double contacts to route the current through the interrupter relay and on to the bell.

The system can be tested by momentarily depressing the V_{NE}-M_{NE} WARN TEST pushbutton located on the pilot's instrument panel. The bell will ring if the circuit is operating properly.

MAIN LANDING GEAR SPEED BRAKE TRIM

When the main landing gear is lowered as a speedbrake, the airplane will tend to pitch up. To provide compensation, the stabilizer trim rate system is automatically switched to the high rate position. This maximum trim rate is available only while the gear is in transit. When the gear is down and locked, and the doors closed, the stabilizer trim rate reverts to the rate demanded by the automatic airspeed switching system.



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Mach/Airspeed Warning System
Figure 7-16

Section 8

AUTOPILOT

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AUTOPILOT

GENERAL AUTOPILOT SYSTEM (PB-20G)

A transistorized automatic flight control system (autopilot) is provided. The system controls attitude, altitude, omnirange course, and instrument landing system (ILS) approach for the airplane. A damper control mode is available to augment yaw stability in manual flight. Some autopilot gains are regulated to changes in altitude and airspeed (see Figure 8-1).

The major components of the autopilot are the controller, a three-axis trim indicator, a glide slope warning light, a three-axis rate control, an air data sensor, an amplifier-computer, a power junction box, an autopilot adapter, three surface servos, a trim servo, two position transmitters, and a vertical gyro transmitter. These components are interconnected with compass and radio circuits to provide all the necessary functions required by the automatic flight control system. Switches are provided to enable selection of either pilots' compass system and the No. 1 or No. 2 radio systems.

Interlock protection of all essential functions required for safe and reliable system operation is provided. All interlocks must be satisfied prior to engagement or the system will not engage. The opening of an interlock while engaged disconnects the system and flashes a warning light.

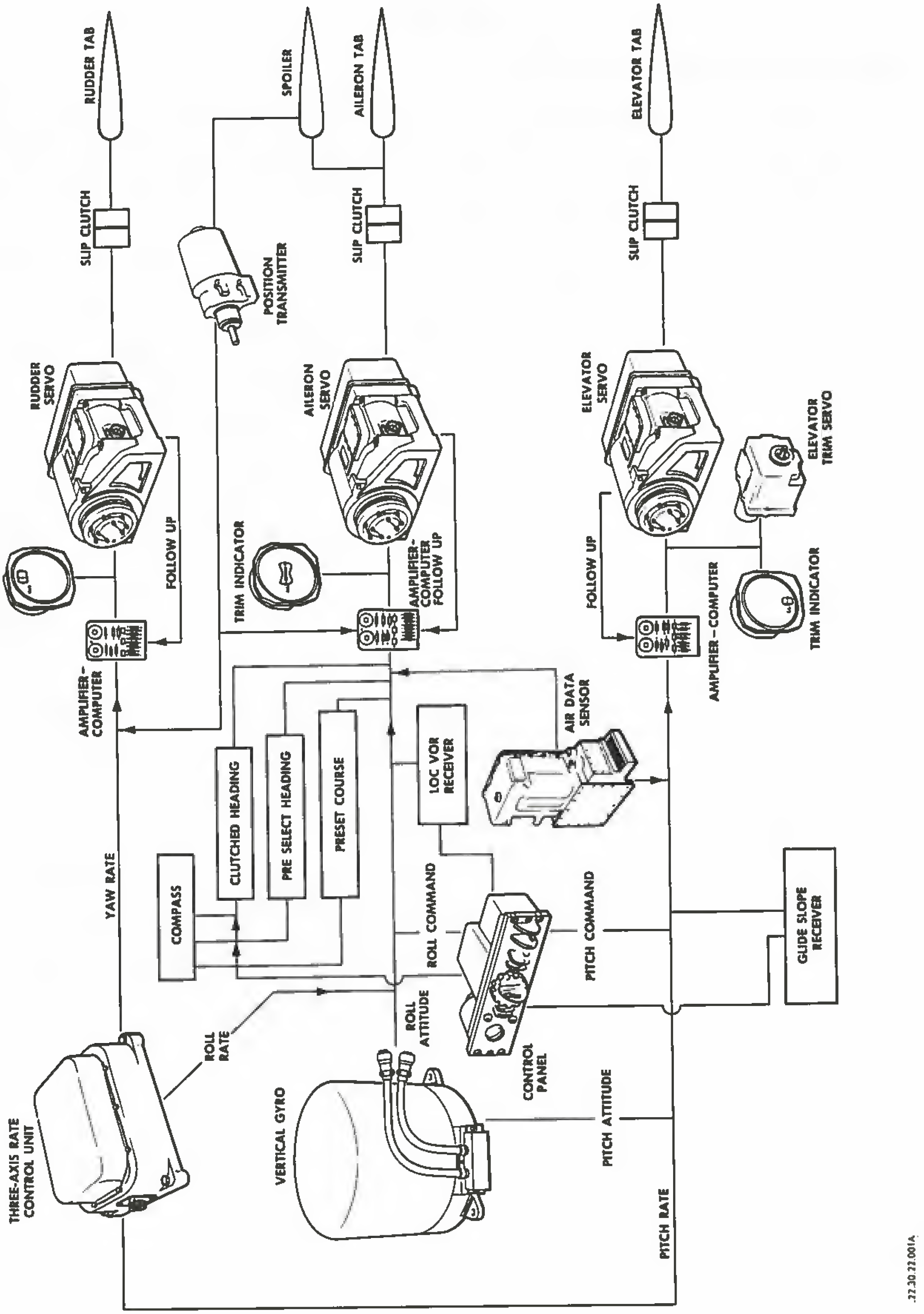
All heading, altitude and radio beam errors are integrated to provide appropriate corrective action. Omnirange data smoothing and synchronizing are performed by electronic computing amplifiers.

The vertical gyro subsystem in the autopilot installation supplies autopilot roll and pitch information and up-elevator to prevent loss of altitude in turns. In addition it furnishes attitude data for radar apparatus and servo controlled indicators.

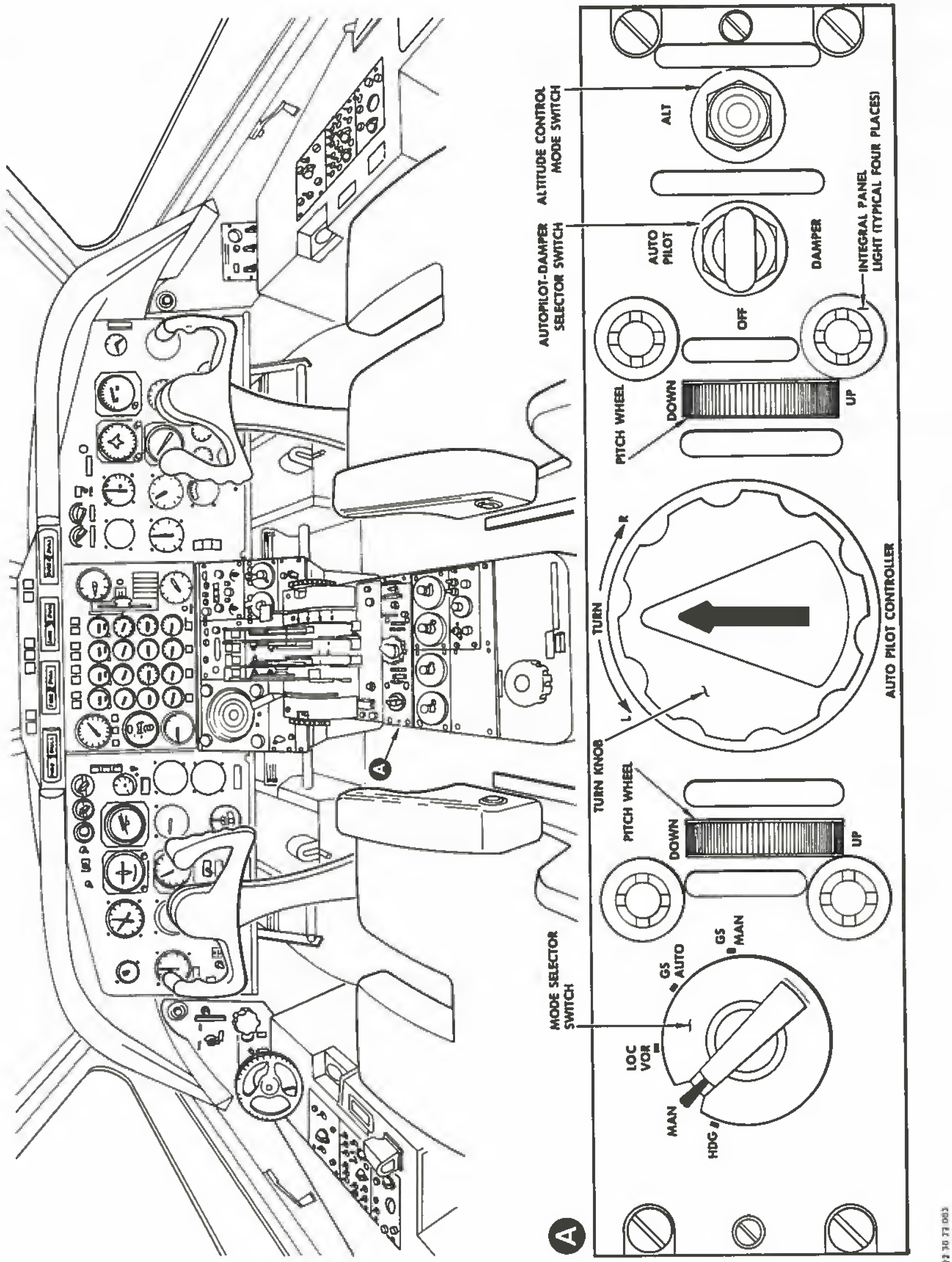
AUTOPILOT CONTROL SYSTEM

Control of the autopilot system is accomplished by the pilot or copilot from the autopilot controller mounted on the pilots' control pedestal. Prior to in-flight autopilot engagement, the airplane should be in trim position. When making system checks on the ground, the controls must be placed in neutral. The display bars on the three axis trim indicators should be in the center-alignment positions before the autopilot is engaged.

Figure 8-2 illustrates the autopilot-damper control panel. Controls for mode switching--turn and pitch maneuvering--and engage switching are mounted on the face of the panel with guards to prevent interference with adjacent settings when the turn control is operated.



Autopilot System Schematic
Figure 8-1



Autopilot Control Panel
Figure 8-2

<u>Autopilot Damper Selector Switch</u>

The autopilot damper selector switch engages either the autopilot system or the damper system function and is a three-position AUTOPILOT-OFF-DAMPER switch. The selective positions of the switch are solenoid-held and if all interlocks are not correct, the switch will return to the OFF position automatically.

<u>Altitude Control Mode Switch</u>

The altitude control switch engages the altitude-hold circuit in conjunction with the autopilot. This switch also includes a solenoid lock-on position. The altitude switch automatically returns to off, when the autopilot is disconnected, and can be engaged only when the autopilot system is engaged and all interlocks are satisfied. The switch also returns to off when operating in the GS/MANUAL position or upon interception of the glide slope beam when in GS/AUTO position.

<u>Turn Knob and Pitch Wheels</u>

The turn control knob and pitch control wheels are linked to the arms of signal potentiometers that are fed from signal transformers. The zero position of each control is the center, and directional movement either way from center increases the signal strength proportionally to provide greater control surface movement. Stabilizer-elevator trim relations are maintained by different signal levels. The signal input to the stabilizer trim servo is parallel to the signal input to the elevator-servo. However, the trim servo operates with higher signal voltage than the elevator servo. When the signal voltage becomes insufficient to drive the elevator servo, the stabilizer trim servo continues to operate to unload the elevator servo by removing the aerodynamic load on the elevator controls.

Two integral microswitches are actuated by the turn knob, one to prevent engagement of the system unless the knob is in center detent and the other to return the mode selector to MAN position when the turn knob is rotated out of center detent during autopilot operation.

<u>MODE SELECTOR SWITCH</u>

The mode selector switch is a rotary type used to select a particular mode of operation. Selective positions are MAN, HDG, LOC/VOR, GS/AUTO and GS/MAN. All positions are solenoid held, except the manual position. If the interlock circuitry is correct when a particular mode is selected, the solenoid will hold the switch in that position. Any time one of the interlocks is not satisfied, the switch will return to the MAN position.

Clutched Heading Control-Manual Mode

The autopilot system will respond to the compass heading whenever the mode selector switch is in the MAN position and the autopilot system is engaged. In this mode, the controller turn knob should be used for all changes in heading.

Preset Heading Control Mode

To follow a preset heading, set the heading selector knob on the course deviation indicator to the desired heading. Place the mode selector switch to HDG position. At any time that the controller knob is moved from the center detent position, the mode selector switch will automatically return to the MAN position.

Omnirange Control Mode

To obtain an omnirange function, the LOC/VOR receiver must be tuned to the correct omnirange station and the COURSE setting on the course deviation indicator must be turned to the desired course radial. Placing the mode selector switch in the LOC/VOR position commands the airplane to approach the beam at a prescribed intercept angle.

Localizer Control Mode

To engage a localizer beam, the radio must be tuned to the desired LOC frequency and the inbound runway heading must be set on the present course (in the CDI window). Placing the mode selector switch in LOC/VOR or GS/AUTO position commands the airplane to approach the beam at a prescribed intercept angle.

NOTE: The same mode selector switch position (LOC/VOR) is used to engage the localizer and omnirange. Prior to selecting the control mode for any radio function, the receiver must be tuned to the proper station frequency.

Glide Slope-Auto Mode

When the localizer beam is captured and before the glide slope intercept is made, the mode selector switch must be placed in the GS/AUTO position. The glide slope warning light on the center instrument panel will illuminate, indicating readiness for automatic glide slope engagement at the intercept point. When the glide slope beam is automatically engaged, the warning light will extinguish.

Glide Slope-Manual Mode

To engage the glide slope manually (pilot flight control), the mode selector switch must be placed in the GS/MAN position just prior to intercept.

Manual Take-Over

At any time that manual control of the airplane is desired, the autopilot disconnect switch on the control wheel must be depressed. This switch returns the airplane to manual control, in a trim condition, by disengaging the autopilot system.

CAUTION: DO NOT DISENGAGE THE AUTOPILOT UNLESS AIRPLANE IS IN TRIM. FAILURE TO HAVE AIRPLANE IN A TRIM CONDITION CAN RESULT IN UNDUE MOVEMENT OF CONTROL SURFACES.

YAW DAMPER SYSTEM OPERATION

The damper system augments airplane stability in yaw during manual control. Placing the autopilot-damper switch in damper position engages the yaw damper system. Only the rudder is placed under automatic control; elevators and ailerons are not affected. Pilots can override the automatic rudder function by applying sufficient force on the applicable rudder pedal. The damper system is disengaged by pressing the autopilot disconnect switch on either control column. Disconnecting automatically returns the autopilot-damper selector switch to the OFF position.

THREE AXIS TRIM INDICATOR

A three axis trim indicator is located on the left side of the pilots' engine instrument panel. The unit is a visual indicator of trim condition when operating on autopilot. Continued displacement of the trim bars from the "floating" center positions is an indication of an out of trim condition requiring manual trim (rudder and aileron) correction. Pitch trim is automatically provided by the trim servo.

CAUTION: THE CONTROL WHEEL SHOULD BE FIRMLY HELD BY PILOT OR COPILOT BEFORE DISENGAGING AUTOPILOT TO CORRECT AN OUT OF TRIM CONDITION.

CONTROL SURFACE SERVOS

Three control surface servos are provided in the autopilot system, one each for rudder, aileron and elevator. These units position the control surface flight tabs in response to electrical signals from the autopilot system. A separate servo is provided for the horizontal stabilizer control system.

AUTOPILOT DISCONNECT WARNING LIGHT

A warning light is located next to the autopilot trim indicator on the pilots' engine instrument panel. The AUTOPILOT DISENGAGED light flashes intermittently when the autopilot system disengages due to a malfunction or when the autopilot-damper switch on the controller panel is moved to OFF position. A steady illumination indicates loss of airplane 28-volt dc power to the autopilot system.

The warning light does not illuminate when autopilot is disengaged by using control wheel buttons. The warning light can be extinguished by pressing the disengage switch on either control wheel.

ELEVATOR OUT OF TRIM WARNING LIGHT

In the event that a sustained out of trim condition exists either in flight or when the landing flaps are lowered more than 2 to 4 degrees, the ELEVATOR OUT OF TRIM warning light, located next to the autopilot disengaged light, will illuminate.

GLIDE SLOPE WARNING LIGHT

A glide slope warning light is located next to the elevator out of trim light and operates in connection with the GS/AUTO position of the mode selector switch. In GS/AUTO position, after proper frequency selection, the light illuminates. Upon intercepting the glide slope beam, the light will extinguish,

AUTOPILOT ELECTRICAL POWER SOURCES

The autopilot system converts airplane electrical power to various ac and dc voltages for autopilot use. Primary input power is three-phase, 115/200-volt ac, 400 cps power from the No. 3 essential ac bus. For detailed information, consult the WIRING DIAGRAM MANUAL.

Warning Light System

Warning lights and warning light control power is from the 28-volt dc emergency bus.

CIRCUIT PROTECTION

A circuit breaker protects each phase of the three-phase electrical power supply to the autopilot. A circuit breaker protects the single-phase supply to the vertical gyro. A fuse protects the 28-volt dc supply to the warning light system. These circuit breakers and fuse are located on the flight compartment main circuit breaker panel.

Three fuses are located on the front of the autopilot power junction box to protect the airplane power input to the autopilot system. A fuse in the same location protects the 27.5-volt dc output power to the servo electromagnetic clutches. These fuses are located on equipment installed in the electrical compartment and are not accessible during normal flight operations.

Section 9

POWER PLANT

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POWER PLANT

ENGINES

Propulsion power is provided by four pod-mounted General Electric CJ805-3 axial-flow turbojet engines. Each complete engine weighs approximately 3642 pounds and is rated in the ten thousand pound thrust class. Special features of the engine include: one major rotating compressor-turbine element, a cannular-type combustion system, a three-stage turbine rotor, variable compressor stator blades, and variable air inlet guide vanes (see Figures 9-1, 9-2 and 9-3). A thrust reverser is mounted on the aft end of the engine turbine frame and is utilized as.an additional braking device during the landing roll of the airplane. A sound suppressor is installed on the aft face of the thrust reverser and forms the aft portion of the engine exhaust section.

ENGINE DATA

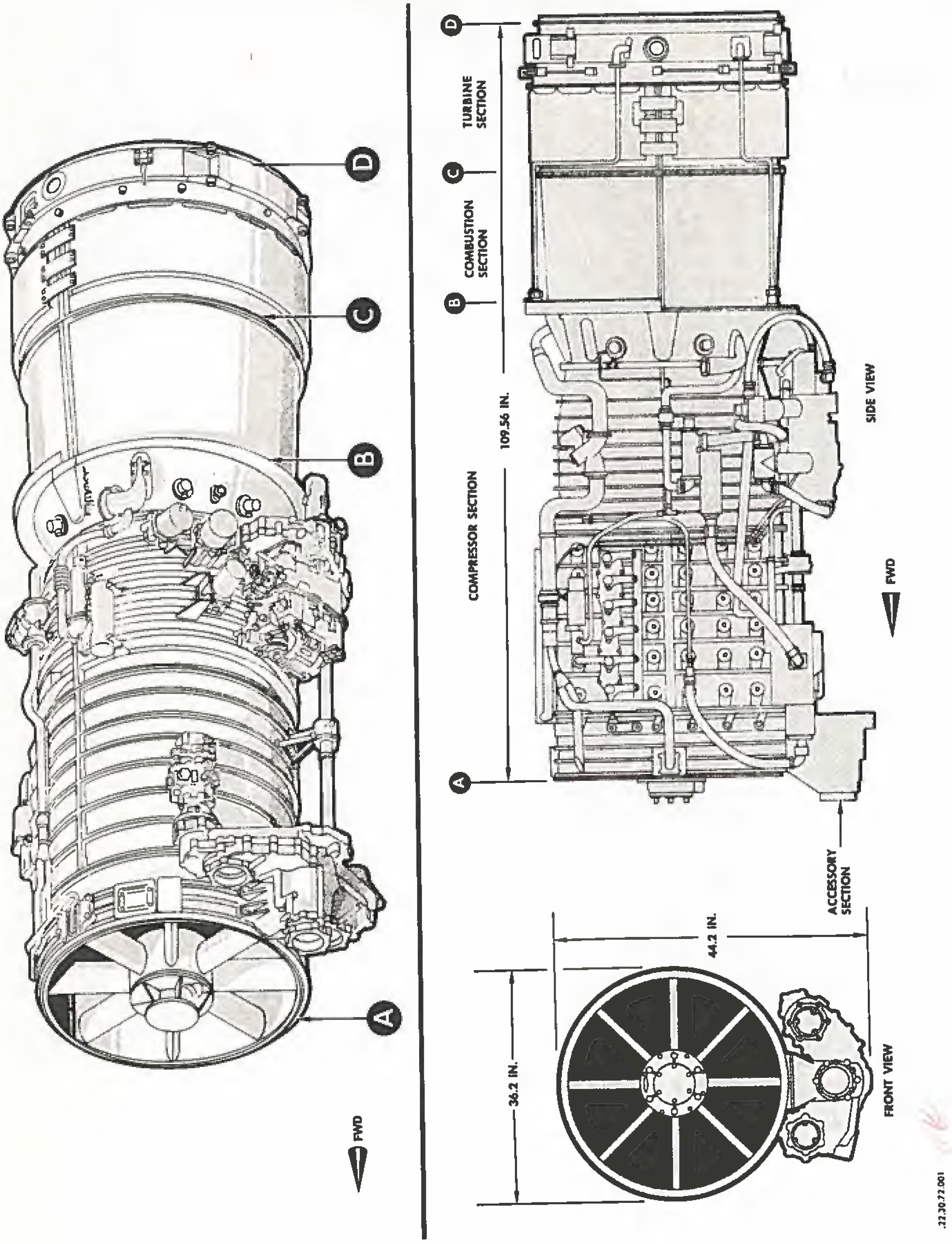
Model Number	CJ805-3
Stages of Compression	17
Compression Ratio	13:1
Fuel Control Device	Hydro-Mechanical
Oil System	Recirculating Type
Ignition System (Dual)	Capacitor-Discharge Type
Anti-Icing System	Hot Air
Engine Speed Indicating	Tach-Generator and Indicator
Exhaust Gas Temperature Indicating (EGT) . . .	Thermocouples-to-Indicator

ENGINE OPERATING CONTROLS

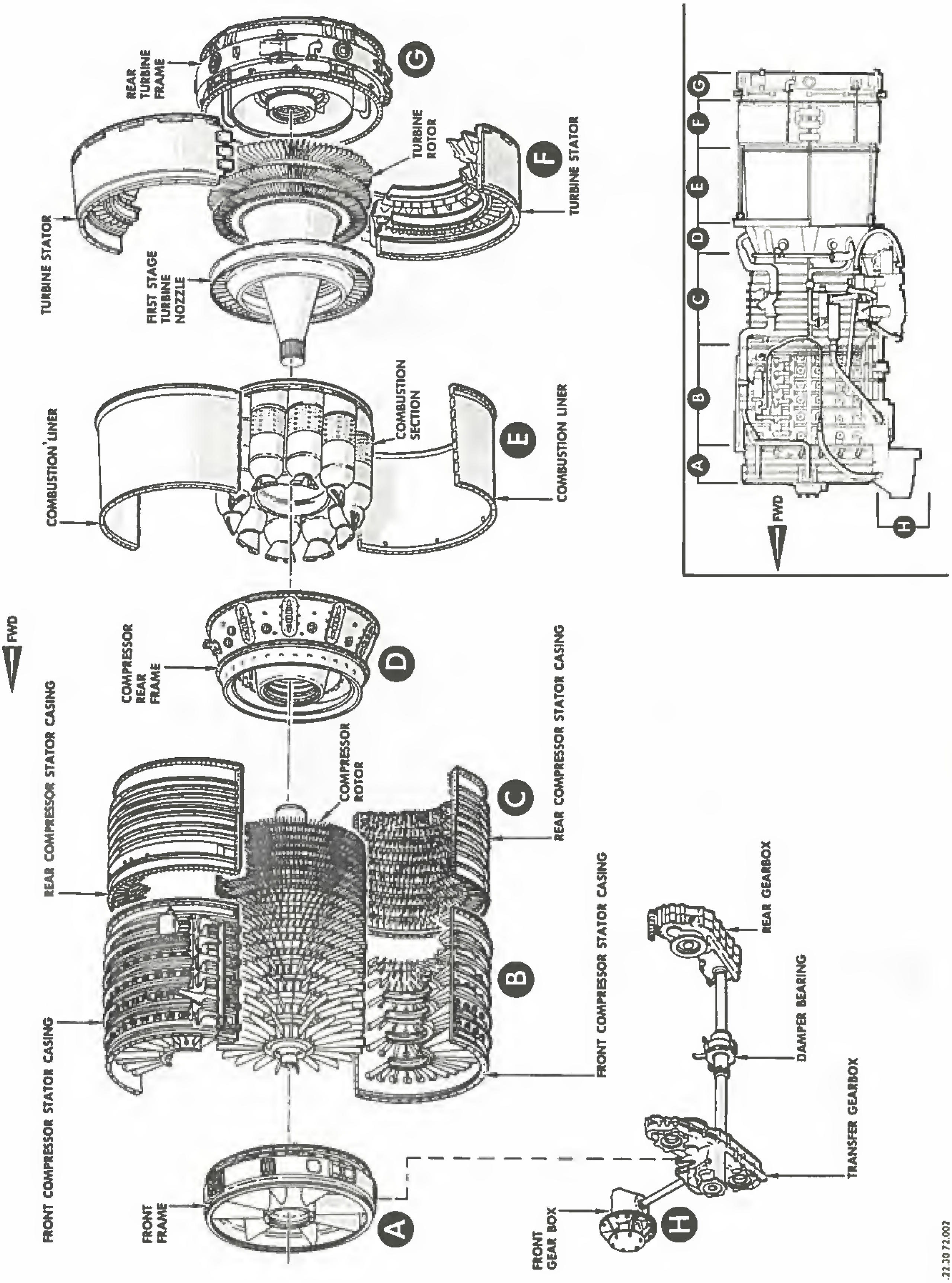
The engine control system consists of a power control lever, a thrust reverse lever, and a fuel shutoff lever for each engine. All controls are mounted on the pilot's pedestal. Individual thrust reverse levers, in the form of an "L", are attached to and forward of the power control lever assemblies between the power control lever handles and the pedestal cover. The thrust reverse levers cannot be used until the power control levers are in the IDLE position (see Figure 9-4).

Power Control Levers

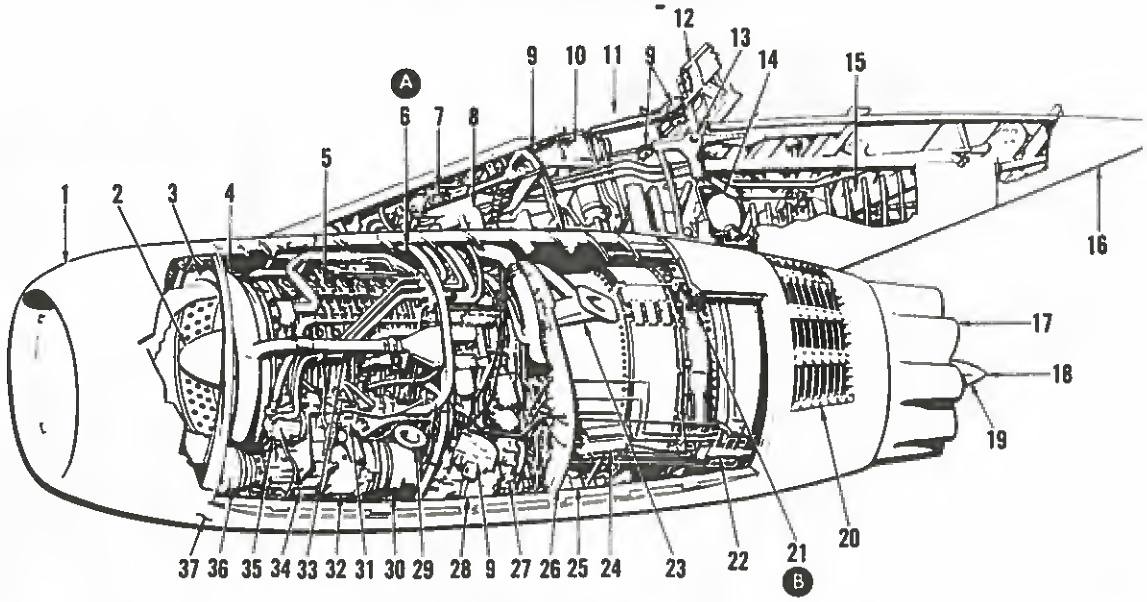
The position of the power control levers during takeoff, cruise and landing, is directly related to the engine fuel scheduling and affects the percent rpm. The power control lever quadrants are marked IDEL POWER, MAX CRUISE, MAX CONT, and TAKEOFF POWER. Cams in the power control lever mechanisms actuate micro-switches to give warning by horn and lights if the slaps, stabilizer trim and landing gear are not in their correct positions at critical times during takeoff and landing.



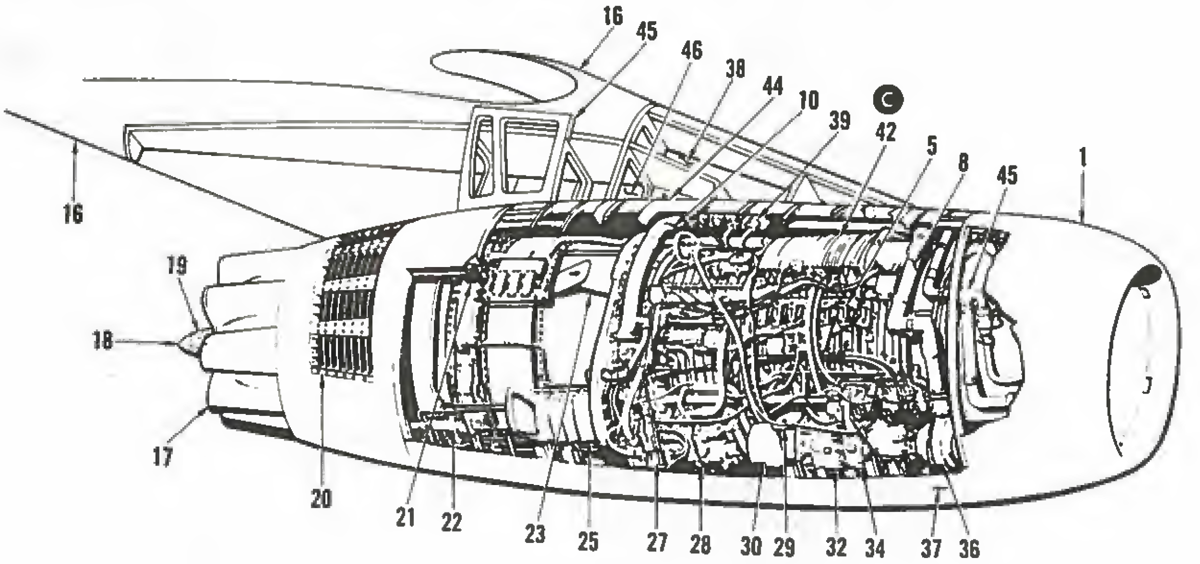
Engine Sections
Figure 9-1



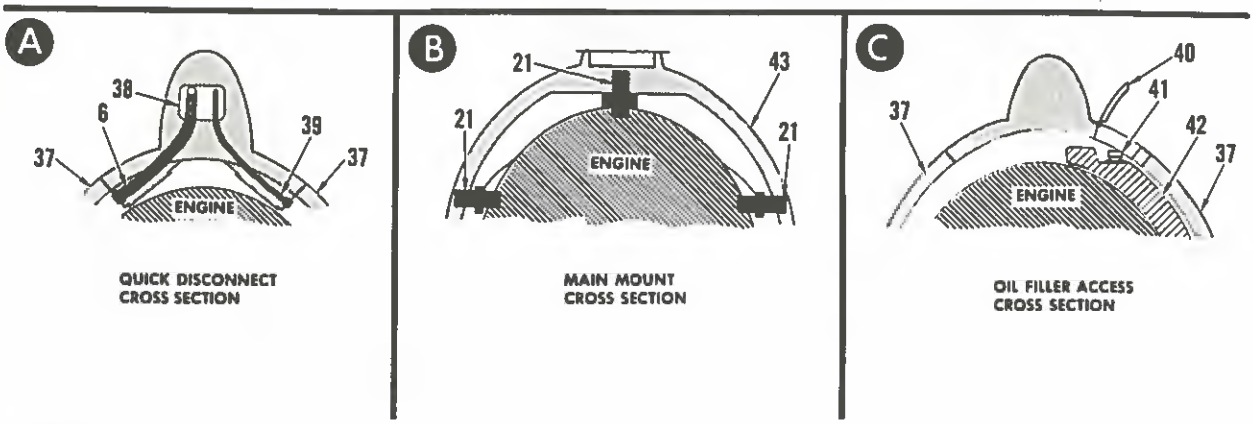
Engine - Exploded View
Figure 9-2



POWER PACKAGE-LH SIDE



POWER PACKAGE-RH SIDE



QUICK DISCONNECT CROSS SECTION

MAIN MOUNT CROSS SECTION

OIL FILLER ACCESS CROSS SECTION

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Power Package
Figure 9-3 (Sheet 1 of 2)

POWER PACKAGE KEY LIST

1. NOSE COWL
2. ENGINE HUB FAIRING
3. ENGINE MOUNTED AIR INLET STUB DUCT
4. FORWARD ENGINE MOUNT
5. VARIABLE STATOR CONTROLS (TYPICAL)
6. FLUID LINES DISCONNECT (LH SIDE)
7. FUEL AND HYDRAULIC LINES
8. ENGINE STARTING AIR DUCT
9. ENGINE FUEL CONTROLS
10. BLEED AIR DUCT
11. PYLON-TO-WING LINK
12. FRONT SPAR
13. MAIN MOUNT CONNECTION TO FRONT SPAR (2)
14. FIRE EXTINGUISHING AGENT CONTAINER
15. AFT DIAGONAL STRUT
16. PYLON FAIRING
17. SOUND SUPRESSOR
18. EXHAUST PLUG
19. EXHAUST TRIM TAB (TYPICAL)
20. THRUST REVERSER CASCADES
21. MAIN ENGINE MOUNT (3)
22. THRUST REVERSER ACTUATOR (2)
23. COMPRESSOR BLEED DISCHARGE DUCT (2)
24. FUEL DRAIN COLLECTOR TANK
25. ENGINE ACCESSORY SECTION COOLING AIR EXIT DUCT
26. VERTICAL FIREWALL SKIRT
27. REAR GEAR BOX
28. FUEL CONTROL UNIT
29. GENERATOR COOLING (INTAKE LH, EXHAUST RH)
30. GENERATOR
31. ENGINE FUEL PUMP
32. CSD (CONSTANT SPEED DRIVE) UNIT
33. CSD AIR-OIL COOLER, VALVE, AND DUCT
34. TRANSFER GEARBOX
35. HYDRAULIC PUMP
36. ENGINE AIRTURBINE STARTER
37. POD DOOR
38. DRAFT SEAL
39. ELECTRICAL QUICK DISCONNECT (RH SIDE)
40. OIL FILLER ACCESS DOOR (RH TYPICAL)
41. ENGINE AND CSD UNIT OIL FILLER CAPS
42. ENGINE AND CSD UNIT OIL TANK
43. MAIN MOUNT STRUCTURE BOX
44. BLEED AIR PLENUN CHAMBER COOLING AIR DUCT
45. NOSE COWL DUCT LIP ANTI-ICE AND VORTEX DESTROYER DUCT
46. BLEED AIR PLENUM CHAMBER

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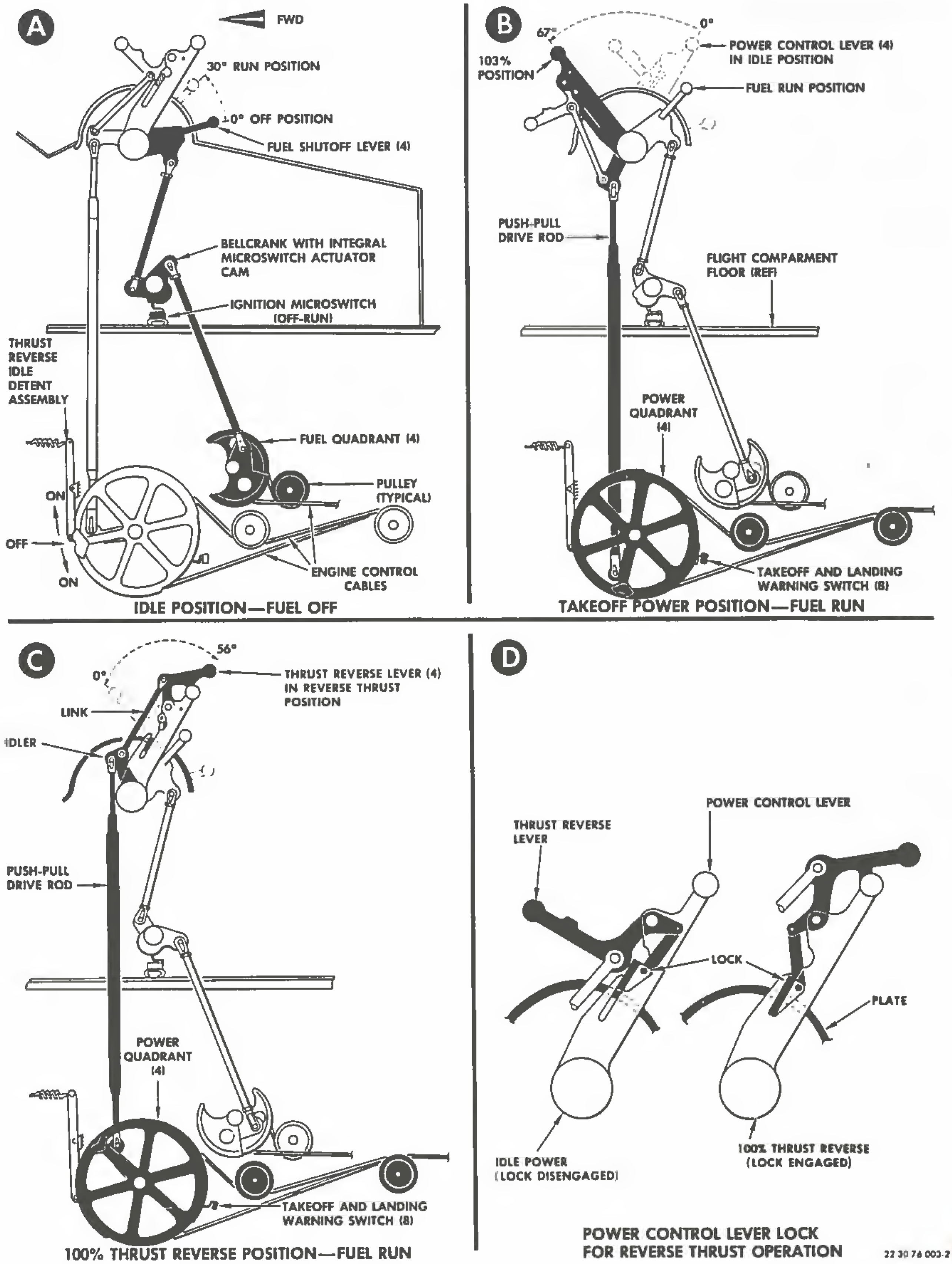


Engine Control Levers
Figure 9-4

Fuel Shutoff Levers

The fuel shutoff levers open the fuel control valves in the fuel control unit and also actuate the engine ignition switches to the "on" position. When the engine speed reaches 3500 rpm, the ignition switch circuit is automatically opened and the starter is disengaged. The fuel shutoff lever quadrants are markes OFF and RUN. The fuel shutoff lever must be pulled outward slightly to release from the OFF position. A similar index is used at RUN position.

Thrust Reverse Levers

The short side of the "L" shaped thrust reverse lever is attached to, and has its pivot point on, the upper section of each power control lever. The thrust reverse levers are locked against operation until the power control levers are in idle position. With the power control lever in idle position, the thrust reverse levers can be moved up and aft. This movement initiates the thrust reverser "clam shell" action and applies increasing engine thrust as lever movement is continued. A "lock-out" safety feature prevents thrust application in the reverse mode on any engine whose thrust reverser system has failed.

ENGINE AIR SYSTEMS

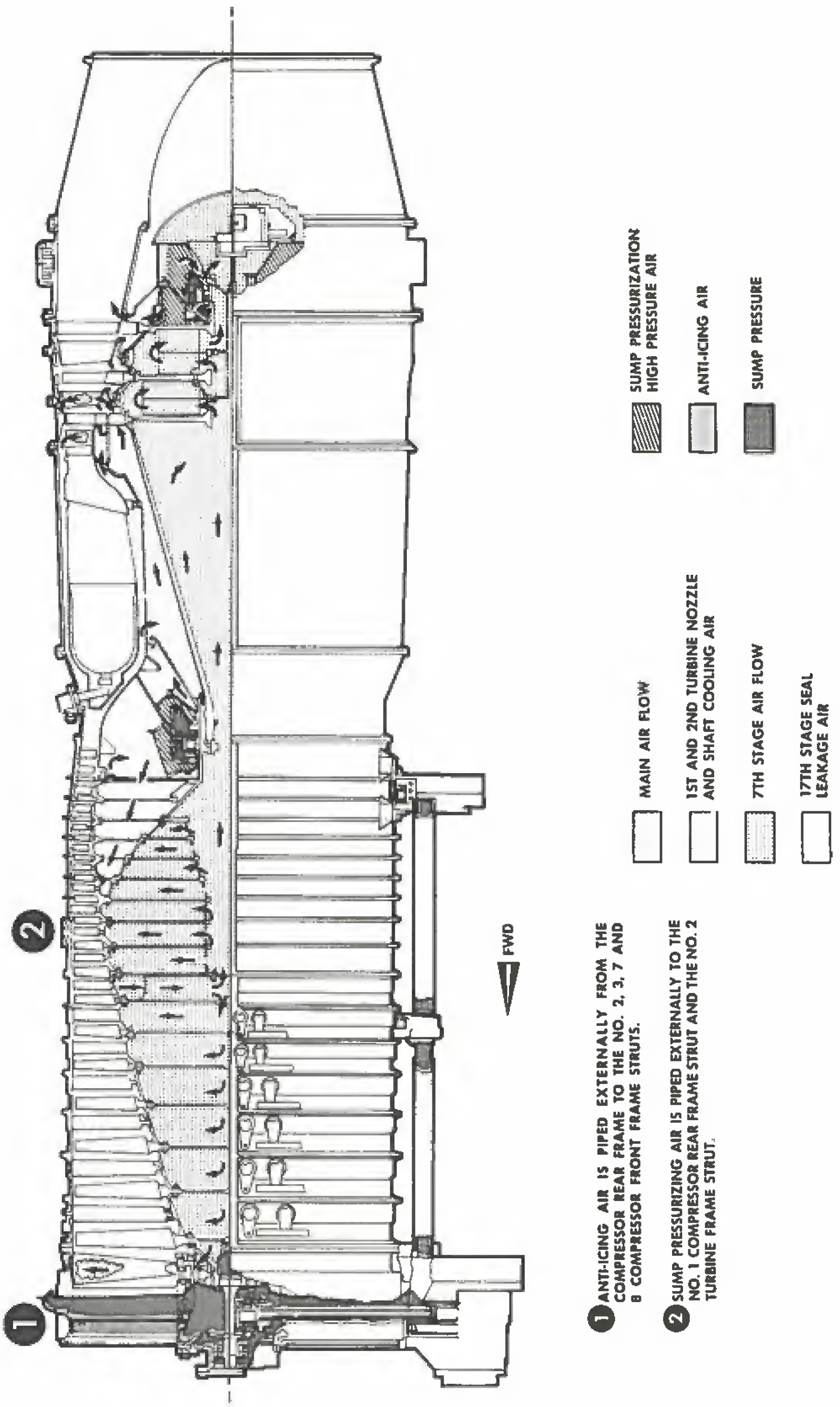
No manual controls are provided for operation of the various engine air systems; however, for purposes of operational information, the air streams can be separated as follows: basic engine air, turbine cooling air, sump pressurizing air system, fuel heater air, anti-icing air system, engine cooling air system and the engine air inlet vortex destroyer (see Figure 9-5).

Basic Engine Air

Air enters the engine at the front frame and passes through seventeen stages of compressor rotor blades and compressor stator vanes. Each stage of blades compresses the air, but not in equal proportions. The exit passages from the compressor section are designed to control the velocity of the air as it enters the combustion section.

The inlet guide vanes and the first six stages of stator vanes are automatically varied. These variable vanes enable the engine to be accelerated and decelerated rapidly without stalling, and to maintain high performance characteristics throughout its operating range.

Air from the compressor section flows into the combustion section. A portion of this air is mixed with fuel and burned. The remaining air blankets the inner and outer casings to provide necessary cooling. The burning of the fuel-air mixture in the liners produces exhaust gases of high velocity. These exhaust gases pass through the transition liners where they are transformed from ten individual streams into a single annular flow. The first stage turbine nozzle then directs the exhaust onto the first stage turbine buckets in the turbine section at the proper angle. The turbine section extracts power from



Engine Air Flow
Figure 9-5

the gas stream to drive the compressor rotor and the various engine mounted accessories. This section consists of the turbine rotor, connected to the compressor rotor, the turbine stators and the turbine frame which is bolted to the outer combustion casing. As the gas stream flows into the turbine section, it strikes the first stage turbine rotor blades. It then continues on through the three stages of rotor blades and turbine stator nozzle guide vanes. Power imparted to the turbine rotor turns the compressor rotor and accessory drives. The gas stream then flows into the exhaust section.

The exhaust section consists of the inner cone, the thrust reverser and the sound suppressor. The primary function of the exhaust section is to transform the gas flow, which leaves the turbine section as annular flow, into a single straight flow at the exit area and discharge the gases at the proper velocity to obtain the maximum thrust.

Thrust Reverser

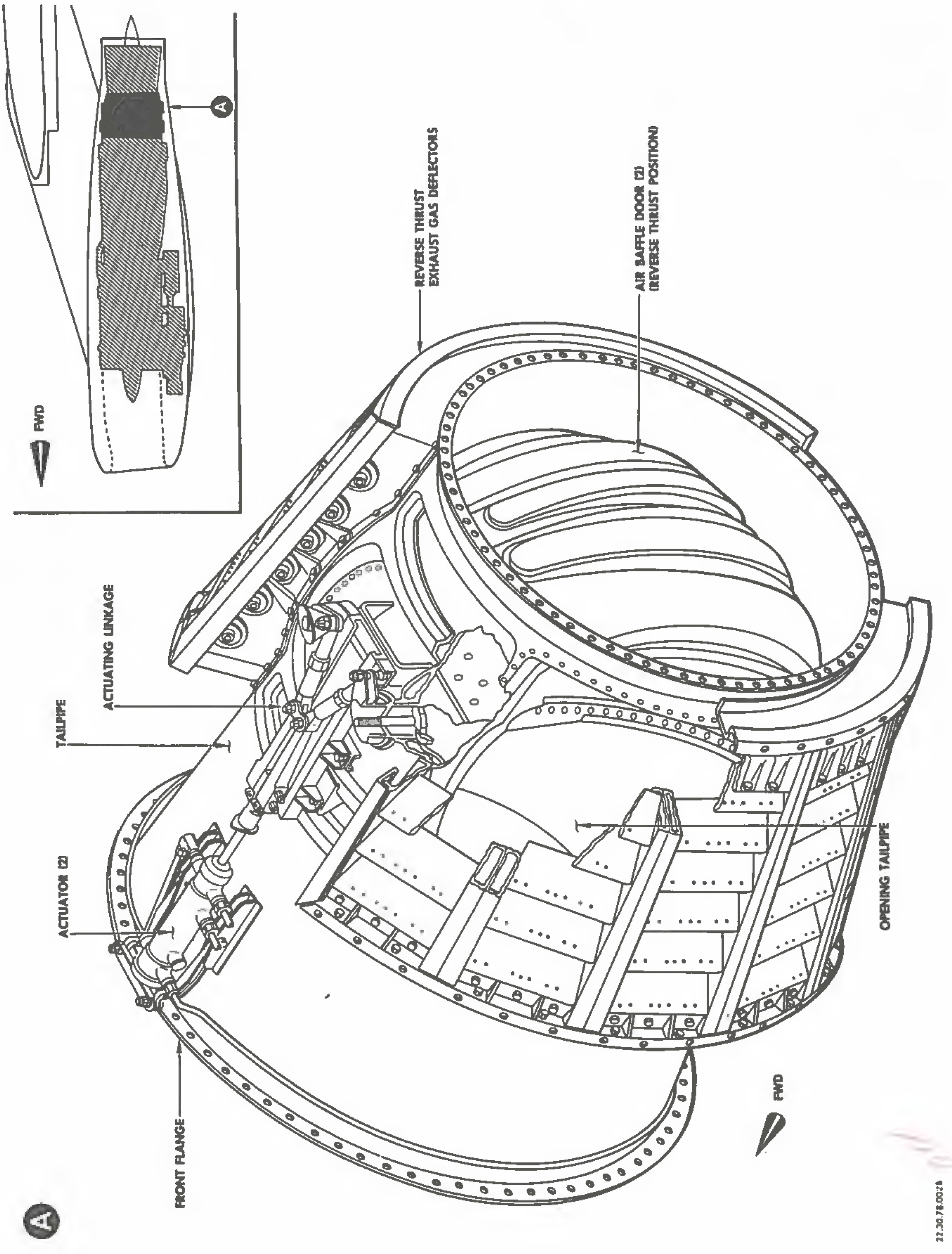
The thrust reverser provides a means of changing the direction of exhaust gas thrust, to create a reverse thrust braking effect, thereby reducing the length of the airplane landing roll after "touchdown". With the reverser in the stowed, or forward thrust position, the exhaust gases flow aft through the reverser section and into the sound suppressor. With the reverser in the reverse thrust position, the gas flow is diverted and discharged from the engine in a forward direction through reverser cascade vanes (see Figure 9-6). The thrust reverser actuating system is hydraulic, utilizing oil from the CSD section of the oil reservoir. For operational details, consult Section 11, OIL SYSTEM.

Sound Suppressor

During forward thrust conditions, the exhaust gases flow through the sound suppressor and are discharged from the engine. The exit configuration of the noise suppressor resembles the petals of a daisy, clustered around a center spike. With this configuration, the gas stream is divided into eight smaller streams. As the eight small streams have a larger mixing surface, a reduction in noise level is achieved.

Turbine Cooling Air

The air used to cool the turbine section components is cool only in comparison with the temperature in the turbine section. The paths which this cooling air takes in the process of cooling the turbine section are as follows: Seventh and ninth stage air, bled internally from the compressor rotor, travels through the inside of the turbine shaft, is directed across each internal face of the wheels by baffles and is discharged into the turbine frame through the stub shaft. Cooling air from the combustion section takes three paths into the turbine section. The outer layer of the combustion cooling air is divided. Part is directed through the first stage turbine nozzle partition and part is directed through the second stage partitions, cooling the rear face of the first stage wheel. The inner layer of combustion cooling air is bled into the area between



Thrust Reverser
Figure 9-6

the inner combustion casing and the turbine shaft through holes in the inner casing. This air is allowed to discharge at the first stage turbine wheel and acts as a coolant for the forward face of the first stage wheel. The No. 3 bearing air seal leakage passes by a baffle in the turbine frame and is then discharged into the exhaust gases. In taking this path it cools the aft face of the third stage turbine wheel and buckets.

Sump Pressurizing Air

The sump pressurizing air system (described in detail in Section 11, OIL SYSTEM) serves a two-fold purpose; it prevents oil leakage past the oil seals and keeps a positive pressure in the sumps. The air system is divided into two parts: high-pressure supply to the No. 1, No. 2, No. 3 bearing areas enclosing the sump; and low-pressure control within the No. 1, No. 2, and No. 3 bearing sumps, the accessory gearboxes and the oil tank.

Fuel-Heating Air

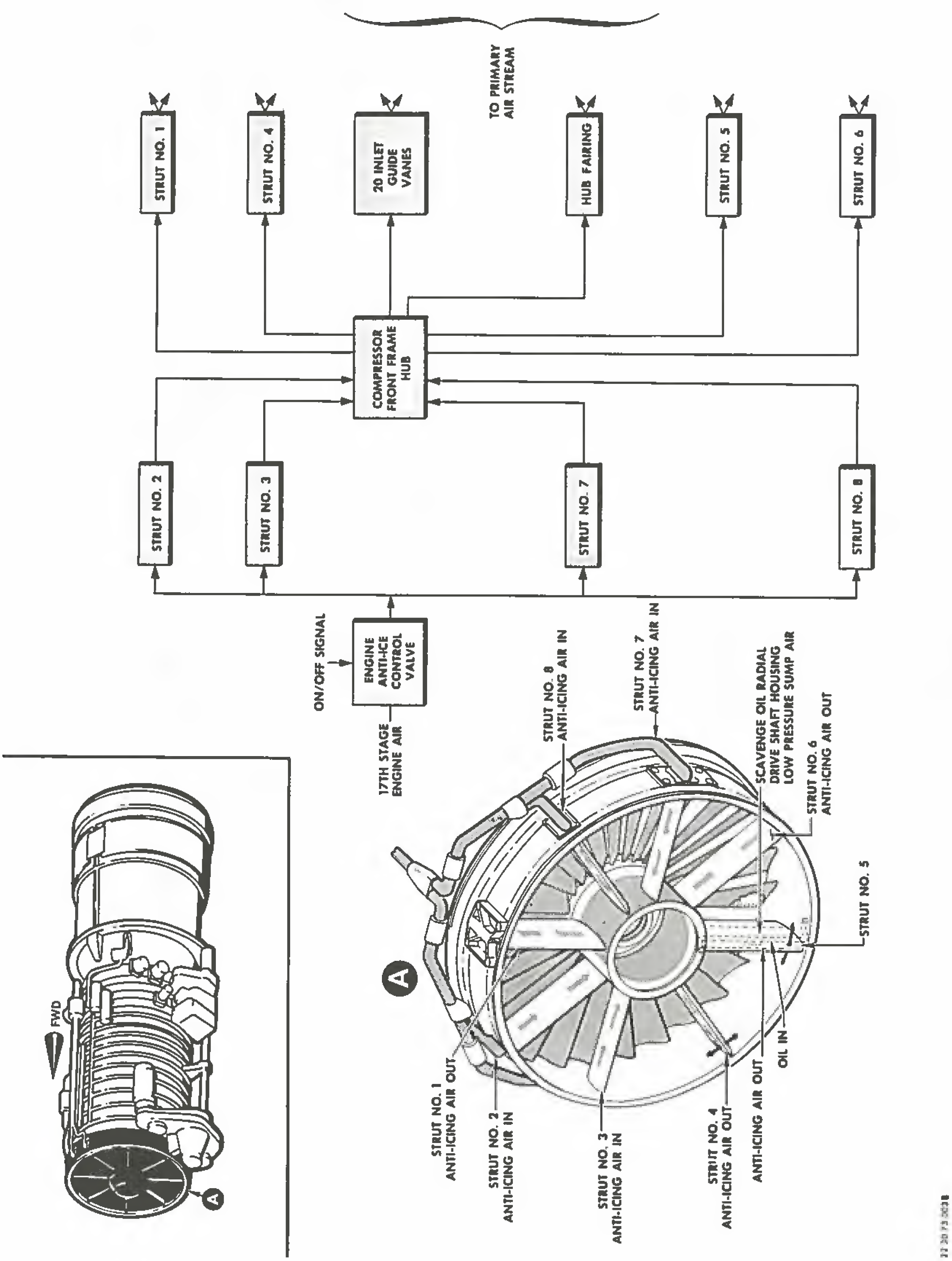
The fuel-heating air system is a means of preventing ice forming in the fuel filter. Seventeenth stage compressor bleed air is taken from the compressor rear frame and routed through a manifold to the fuel heater, an air-to-liquid type heat exchanger. The heater includes an automatic air shutoff valve which controls the flow of air as a function of heater discharge fuel temperature.

Engine Anti-Icing Air

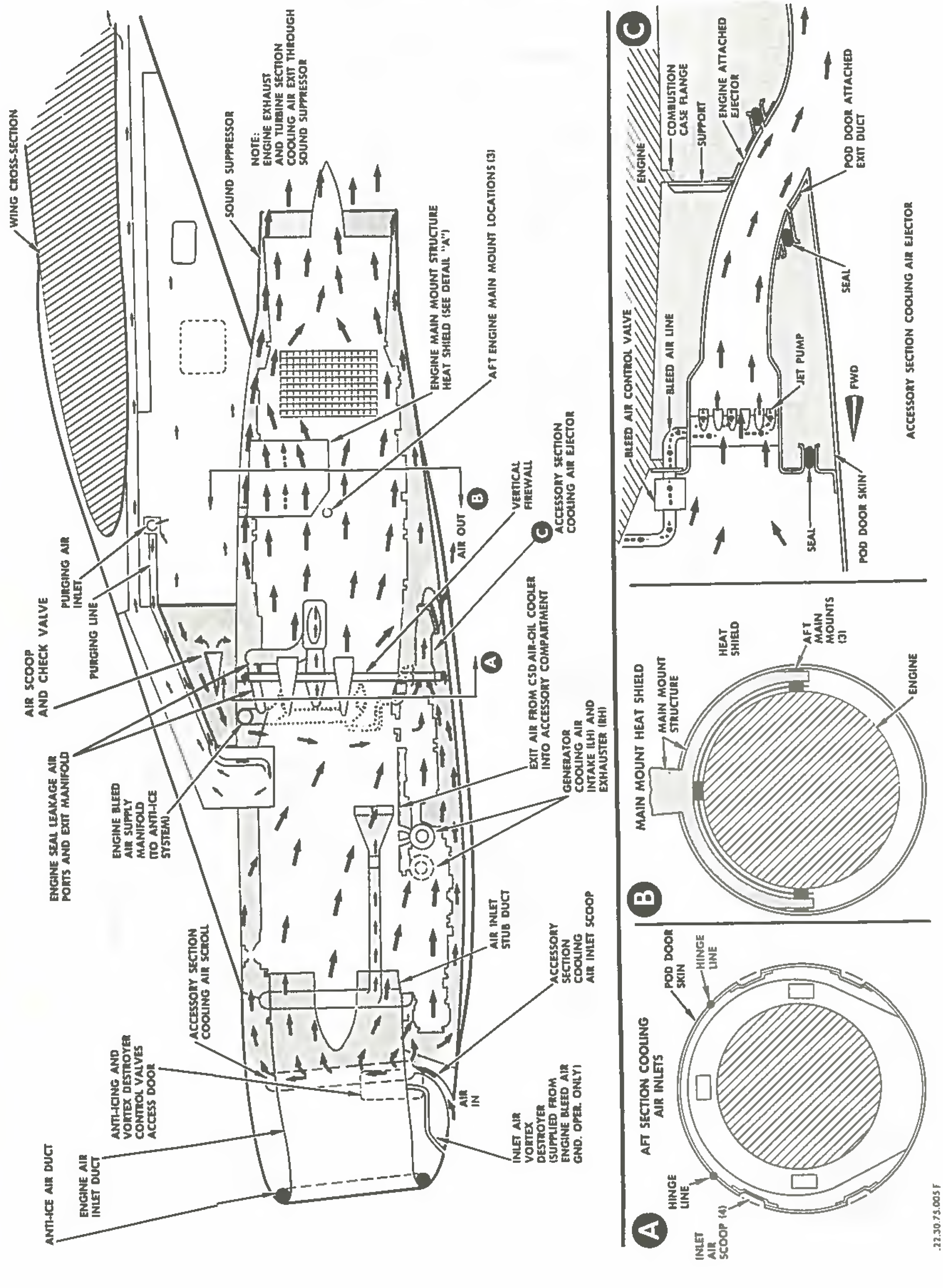
The engine anti-icing system prevents accumulation of ice in the compressor inlet. Anti-icing is accomplished by passing seventeenth stage compressor discharge air from the compressor rear frame through a valve to the front frame struts and inlet guide vanes by means of a manifold. Specifically, the system anti-ices the compressor front frame hub, bullett, and struts, the inlet guide vanes and the nose cowl inlet air lip (see Figure 9-7). The system can be controlled manually by the crew or can be connected to the airplane ice-sensing system for automatic operation.

Engine Cooling Air System

The engine and accessory cooling system consists of the engine accessory compartment cooling air inlet and distribution duct, generator cooling duct, generator cooling air entrance and exit duct, constant speed drive (SD). air-oil cooler duct, engine accessory compartment, cooling air exit duct, engine turbine compartment cooling inlet and distribution ducts and exit, leakage air manifold and exits and the main engine mount heat shield. The cooling system and figuration eliminates the reverse flow of air from the engine compartment areas back into the engine inlet duct, and provides adequate cooling with minimum drag losses for all airplane operational conditions. Cooling air airscoops are so located that a fire cannot enter from any other pod compartment or from any part of the airplane (see Figure 9-8).



Compressor Inlet Anti-Icing Air Flow
Figure 9-7



Engine Pod Cooling Air System
Figure 9-8

Engine Air Inlet Vortex Destroyer

A jet air blast system is installed in the lower leading edge of the nose cowl just aft of the anti-ice inlet section. The jet blast airstream is directed downward from the nose cowl to the ground at the vortex base of the suction pressure that is generated between the ground and the engine air inlet during engine ground operation. The function of the jet blast is to "break-up" forces that occur when a vortex forms, thereby minimizing ingestion of foreign material into the engine during ground engine operation. The jet blast system uses engine bleed air for its pressure source and is controlled by an automatic shutoff valve in the nose cowl. The shutoff valve is connected to a landing gear switch to prevent vortex destroyer operation while airborne.

IGNITION AND STARTING SYSTEM

The ignition circuit provides the spark necessary to ignite the fuel-air mixture in the combustion section of each engine. Once ignited, the fuel-air mixture continues to burn; therefore, it is necessary to power the ignition circuit only during the engine starting period. The ignition circuit consists of two high energy ignition systems, each system having its own lead, voltage booster and ignitor plug (see Figures 9-9 and 9-10). Ground engine starting is accomplished through the use of individual air turbine starting units on each engine. Each starter incorporates an engaging mechanism to permit automatic engagement and disengagement with the engine, a turbine chamber which houses the turbine blades, and a radial air inlet nozzle. Equipment built into the starter control valves prevents overspeeding of their turbine wheels when the starters disengage from the engine at starter cutoff speed.

In case of a four engine flame-out during flight, or an emergency ground start without ac available, the emergency inverter is used to power the ignition system. A switch on the flight engineer's instrument panel connects the battery powered inverter into the ignition system.

Ignition and Starting Controls

The ignition and starting controls consists of one IGN SEL switch, one NACELLE-MANIFOLD selector switch, and four GROUND/FLIGHT starter switches. All of these switches are located on the pilot's overhead switch panel.

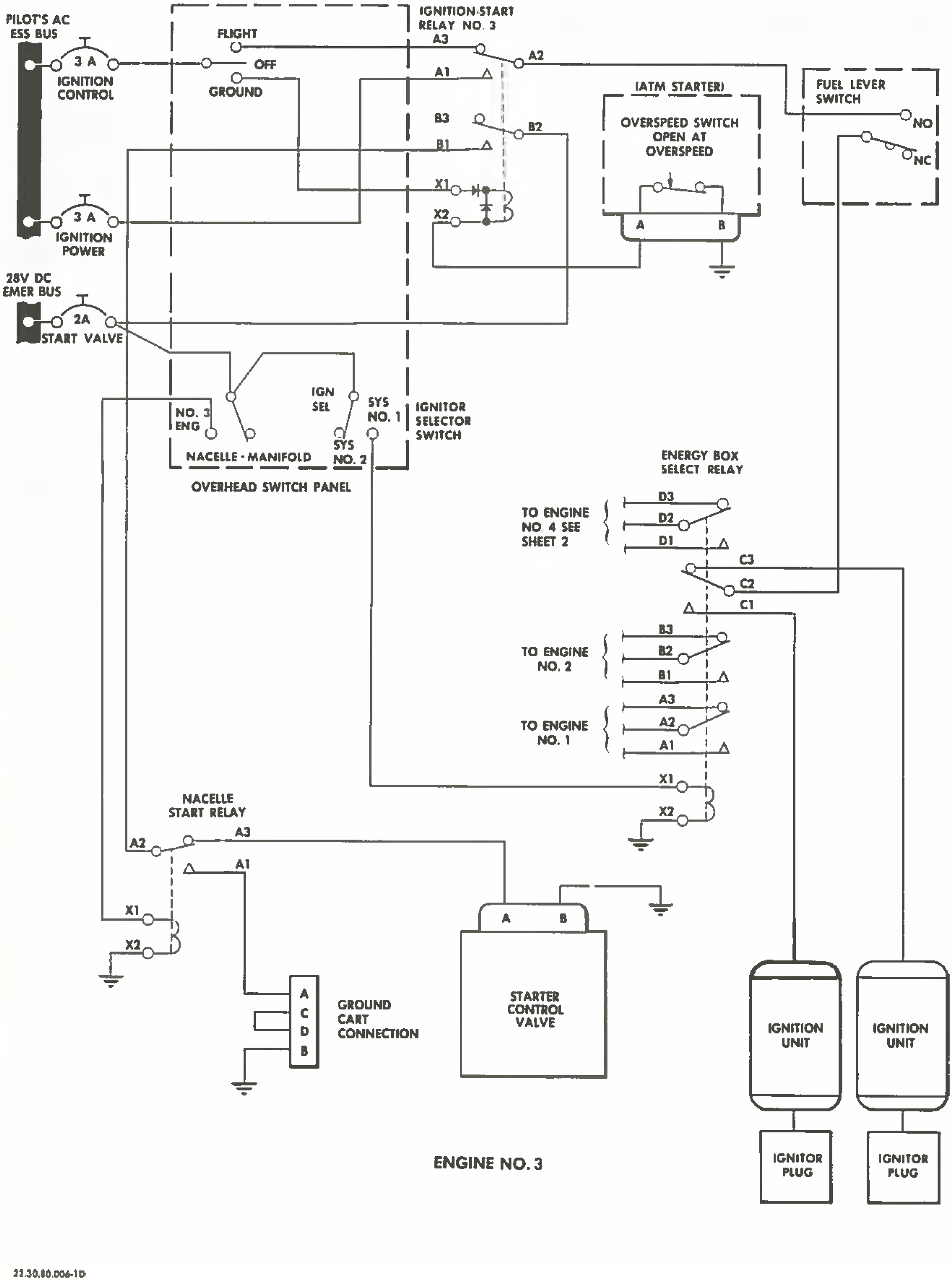
Ignition Selector Switch

The IGN SEL switch is located on the ENGINE STARTER switch panel and enables the pilot to select SYS No. 1 or SYS No. 2.

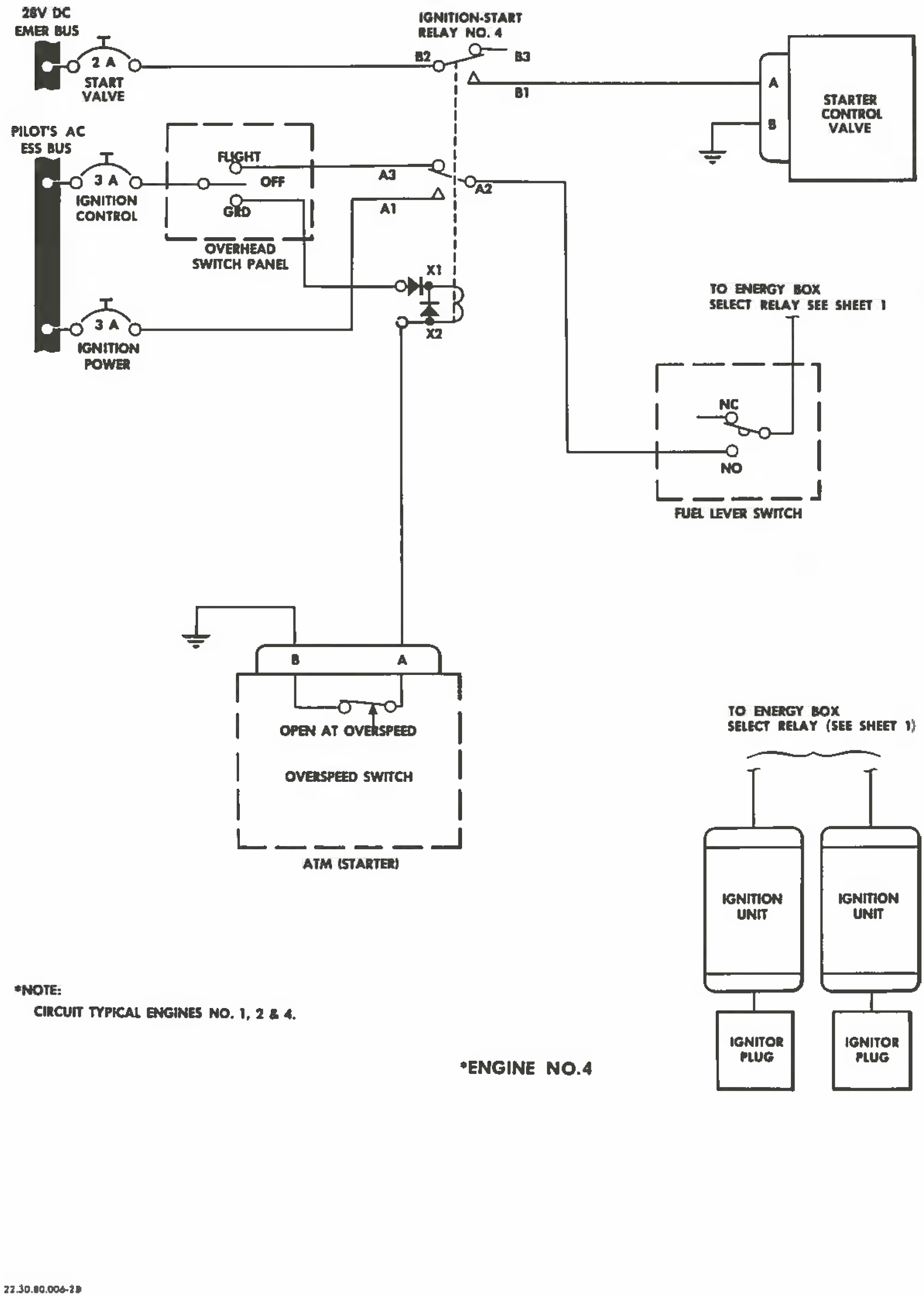
Nacelle-Manifold Selector Switch

Two methods of obtaining air for ground starting are provided:

1. A ground air connection is located on the right underside of the fuselage near the wing front spar. This connection is used to supply air from a ground source through the airplane crossover bleed system manifold to start any engine or engines.



Engine No. 3 Starter and Ignition System Circuit
Figure 9-9



Engine No.'s 1, 2 and 4 Starter
and Ignition System Circuit
Figure 9-10

2. An in-line combustor ground cart engine starting connection which incorporates an integral check valve is installed in the "tee" duct interposed between the starter and starter control valve at engine nacelle No. 3. Electrical connections permit control of the ground cart from a switch in the flight compartment.

The NACELLE-MANIFOLD selector switch allows selection of method 1 or 2 as described above (see Figure 9-11).

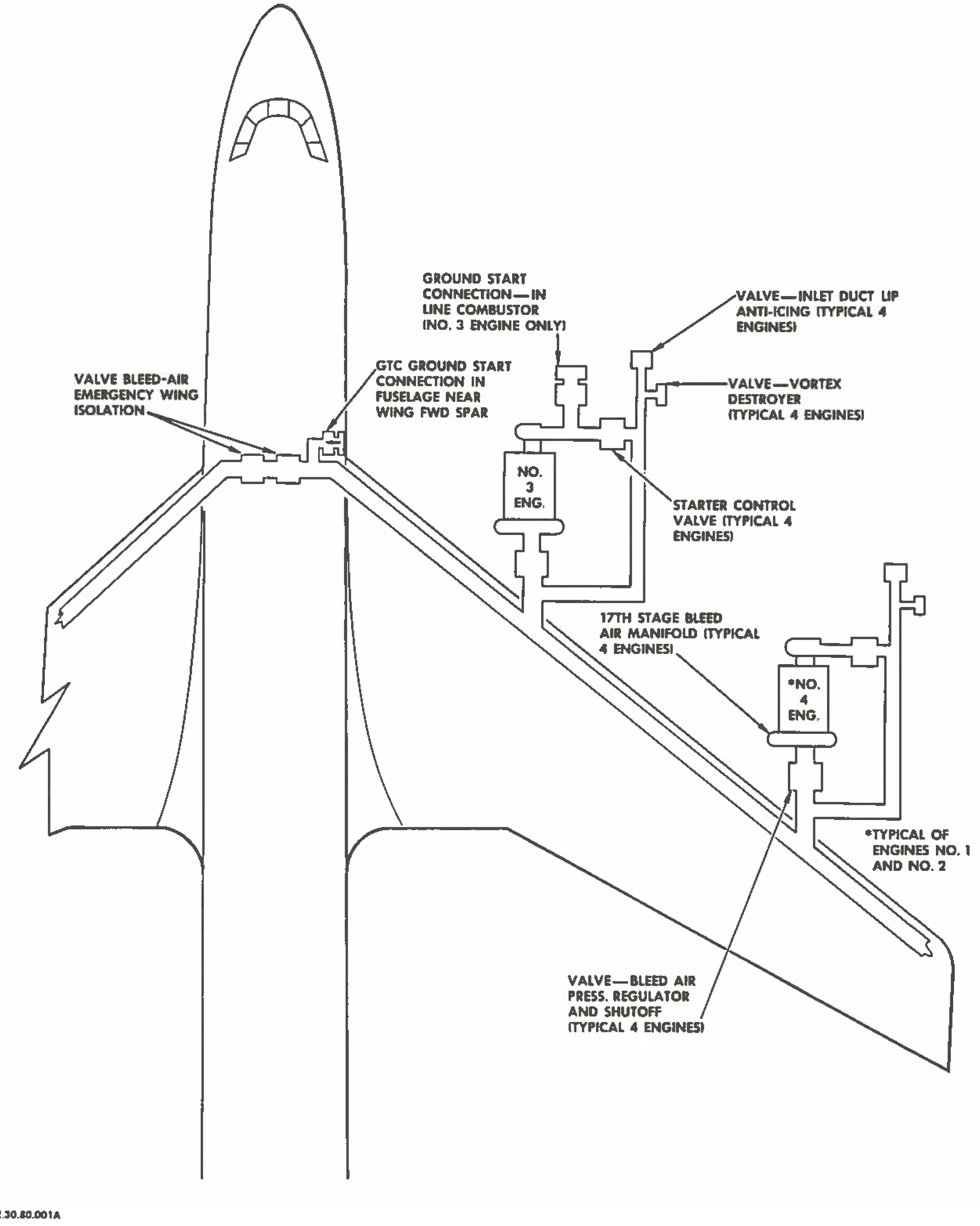
Ground-Flight Starter Switches

The four ground-flight starter switches are three position, GROUND-(off)-FLIGHT, toggle switches. In GROUND position, the switch energizes an ac operated relay that completes its circuit through a centrifugal cutout switch operated by the air-turbine starter. This ac operated relay actuates the air starter control valve, supplying air to the air-turbine starter. At the same time, this switch makes 115-volt ac power available to the ignition switch operated by the fuel shutoff lever. The 115-volt ac power continues through the fuel shutoff lever controlled switch, when in RUN position, to the No. 1 or No. 2 ignition system, depending on the position of the ignition system selector switch. In FLIGHT position, the air-turbine starter is inoperative.

ENGINE INSTRUMENTS

The various engine operation indicating systems are installed on the pilots' engine instrument panel and the flight engineer's panel. Engine anti-icing and bleed air controls and indicators are located in the pilots' overhead switch panel. Engine indicating systems are:

1. Engine speed indicating system (RPM).
2. Engine pressure ratio indicating system (EPR).
3. Exhaust gas temperature indicating system (EGT).
4. Oil system indicating systems: (see Section 11, OIL SYSTEM)
   A. Oil quantity indicating system.
   B. Oil temperature indicating system.
   C. Oil pressure indicating system.
   D. Oil low-pressure warning light system.
5. Fuel indicating systems: (see Section 10, FUEL SYSTEM)
   A. Fuel flow indicating system.
   B. Fuel temperature indicating system.
   C. Fuel pump low-pressure warning light system.
6. Thrust reverser position indicating system.



Engine Starting Air Supply Schematic
Figure 9-11

Engine Speed Indicating System

The engine speed indicating system consists of the tachometer generator and an indicator for each engine. The generator is driven through a gearbox and generates its own voltage for operation of the system. The generated voltage is transmitted to the indicator which displays the signal as an indication of engine rpm in percentage figures.

Exhaust Gas Temperature Indicating System (EGT)

The exhaust gas temperature indicating system incorporates transistorized, null-balancing indicators and dual-loop thermocouple harnesses. One loop of each dual loop thermocouple harness is attached to each side of an engine. The millivolt output of each thermocouple is calibrated to the temperature indicating system's heat range curve, and the resultant voltage is compared to a reference voltage existing in the indicator. The comparison result is displayed in degrees Centigrade on the exhaust gas temperature indicator (EGT).

Engine Pressure Ratio Indicating System

The engine pressure ratio indicating system consists of a transmitter and an indicator for each engine. The transmitter senses pressure from a total pressure pickup in the sound suppressor. This pressure is compared to ram pressure supplied by the copilot's pitot line. The two values are integrated and the results are converted to an electrical signal. The electrical signal is transmitted to the indicator where it is amplified and displayed in pressure ratio readings. A power off warning flag is provided.

Thrust Reverser Position Indicating System

Thrust reverser position indication is accomplished through microswitches which are interlocked with the engine power control levers. The thrust reversers cannot be activated until the power control levers are in IDLE position. Actuation of the thrust levers closes the microswitches, illuminating the blue IN-TRANSIT lights. When the reversers are in full functioning position, amber REV-THRUST lights illuminate. The IN-TRANSIT lights also illuminate during the return of the thrust reversers to normal position, or if the thrust reversers are stuck in-transit.

ENGINE CONTROL AND INDICATION CIRCUIT ELECTRICAL SOURCES

Engine anti-icing air system, pilot's ac essential bus.
Ignition, pilot's ac essential bus.
Starter switch, 28-volt dc emergency bus.
Ground cart-manifold switch, 28-volt dc emergency bus.
Engine speed indicating system, self powered.
Exhaust gas temperature indicating system, pilot's ac essential bus.
Thrust reverser position indicating system, 28-volt dc emergency bus.
Engine pressure ratio indicating system, pilot's ac essential bus.
For detailed information, consult the WIRING DIAGRAM MANUAL.

Section 10

FUEL SYSTEM

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FUEL SYSTEM

GENERAL FUEL SYSTEM

The fuel system includes subsystems designed to supply and control the flow of fuel to the four engines without interruption under all flight conditions. In the supply systems, two main tanks in each wing are divided into four fuel-tight compartments. Two compartments serve as replenishment supply to the two main tanks. Each wing fuel system operates independently of the other. Each main tank contains two electric fuel boost pumps, located in a well, each operated from a different power source, and one hydraulically operated jettison pump. In addition, each inboard main tank well contains a hydraulically operated fuel scavenge pump.

The engine fuel control system regulates the amount of fuel delivered to the engine in relation to the desired rpm, compressor inlet air pressure and temperature, and acceleration and deceleration rate. Fuel from any main tank can be delivered to any or all operating engines.

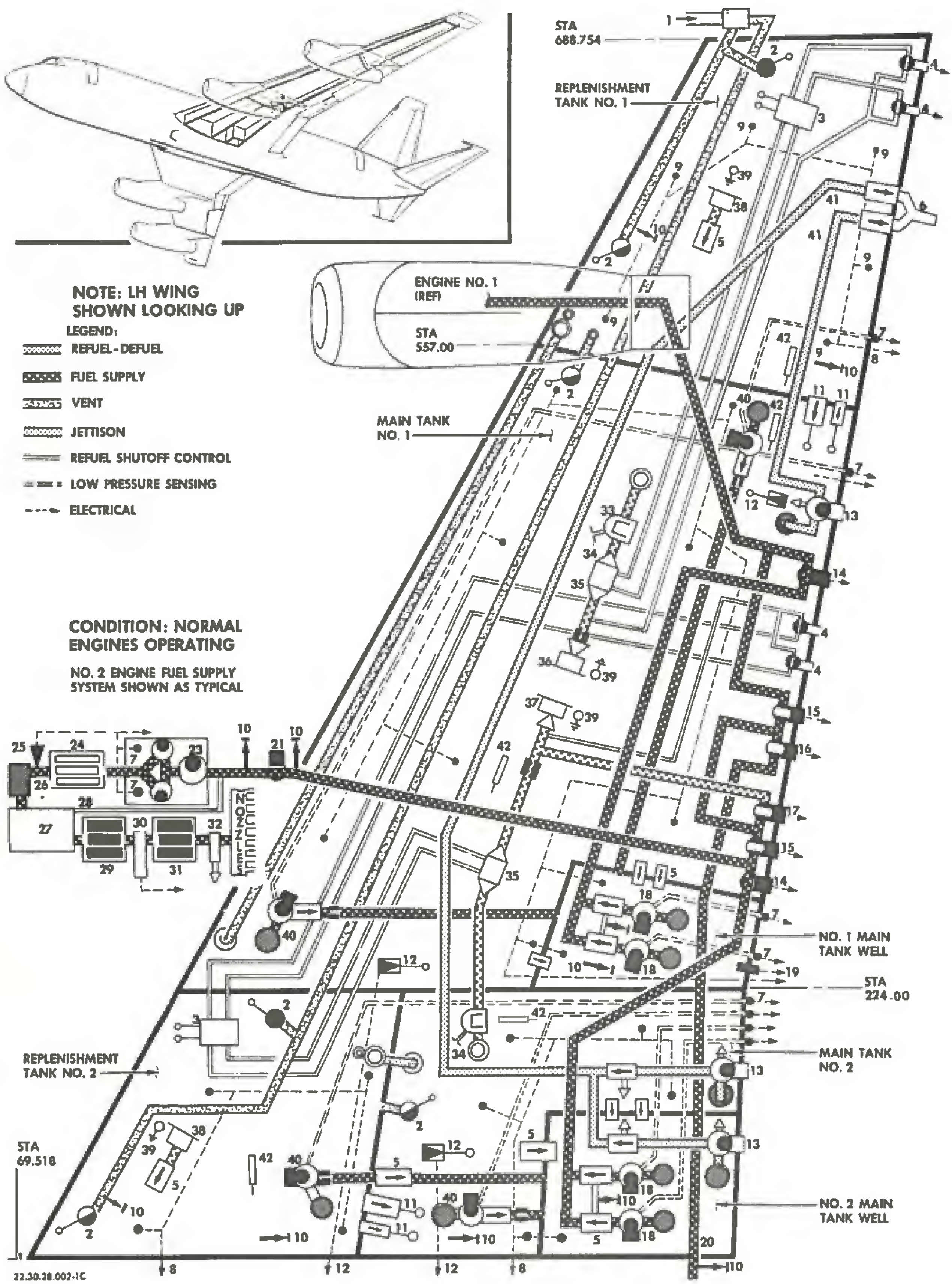
The normal refueling procedure uses underwing pressure refueling adapters. Automatic selection of any quantity up to full tank, exclusive of the minimum allowable expansion space, is possible. Fuel tanks can be filled simultaneously at a rate of 1200 gpm by using all four fuel adapters, or individually at a rate of 300 gpm at 50 psi.

Engine Fuel Requirements (Approximate, depending on various conditions).

Maximum takeoff power and speed based on 103 percent rpm requires a fuel flow of 10,400 pph per engine at sea level, decreasing to 8300 pph at 10,000 feet pressure altitude. Maximum climb power and cruising speed at maximum gross weight (97 percent rpm) requires a fuel flow of 6500 pph per engine at 10,000 feet pressure altitude, decreasing to 2200 pph per engine at 40,000 feet.

FUEL TANKS

Fuel tanks are numbered 1, 2, 3, and 4 from left to right; each tank normally supplies fuel to its correspondingly numbered engine (Figure 10-1). During crossfeed operation, any tank may be used to supply any or all engines. Control combinations are possible to the extent that all four engines can be supplied with 100 percent of maximum cruise fuel flow from any one tank. The tanks, sealants and components are suitable for use with JP-4 fuel, aviation type gasoline, and aromatic blends of JP-1 fuels and kerosenes, provided the blends range from 3 to 30 percent. All tanks have a minimum expansion space equal to two percent of the tank gross volume. Consult the Flight Manual, Section 1, CERTIFICATE LIMITATIONS, for fuel system requirements and limitations.



Fuel System Schematic
Figure 10-1 (Sheet 1 of 2)

SYMBOLS USED:

- PUMP—AC MOTOR
- PUMP—HYDRAULIC MOTOR
- VALVE—DC MOTOR
- VALVE—DC SOLENOID
- BELLMOUTH DOWN
- BELLMOUTH UP
- SCREENED BELLMOUTH DOWN
- SCREENED BELLMOUTH UP
- BELLMOUTH WITH STANDPIPE
- CHECK VALVE
- CHECK VALVE FLOAT OPERATED
- CHECK VALVE WITH LINE DRAIN
- RESTRICTOR—INLINE
- RESTRICTOR—SEAL TYPE
- SWITCH-TRANSFER PUMP FUEL LEVEL FLOAT
- FLUSH DRAIN VALVE EXTERNAL DRAIN
- VENT VALVE—FLOAT ACTUATED
- VENT VALVE—FLOAT ACTUATED WITH PRESSURE RELIEF
- FUEL LOW-PRESSURE WARNING SWITCH
- FUEL PROBES
- LIQUID SIGHT GAGE (DRIP STICK)
- REFUEL SHUTOFF VALVE
- REFUELING PILOT VALVE
- PRESSURE REFUEL ADAPTOR
- GRAVITY REFUEL ADAPTOR AND CHECK VALVE
- GROUND JACK
- REGULATOR—FUEL TANK PRESSURE

NUMERICAL CODE:

1. FLUSH RAM-VENT SCOOP
2. VENT VALVE-FLOAT ACTUATED (2)
3. REFUELING PILOT VALVE (2)
4. STOP-FUEL SOLENOID VALVE (4)
5. CHECK VALVE (TYP)
6. JETTISON NOZZLE
7. FUEL LOW-PRESSURE WARNING SWITCH (10)
8. ELECTRICAL SYSTEM-CONNECTION (4)'TO PROBES
9. FUEL PROBES (TYP)
10. FLUSH DRAIN VALVE (TYP)
11. FLOAT ACTUATED CHECK VALVE (4)
12. FUEL LEVEL FLOAT SWITCH (3)
13. JETTISON PUMP—HYDRAULIC MOTOR OPERATED (3)
14. LINE SHUTOFF VALVE (2)
15. CROSSFEED VALVE (2)
16. EMERGENCY CROSSFEED VALVE
17. DEFUEL VALVE
18. FUEL BOOSTER PUMP (4)
19. TEMPERATURE GAGE (LH WING ONLY)
20. EMERGENCY FUEL CROSSFEED LINE
21. EMERGENCY FUEL SHUTOFF VALVE
22. (DELETED)
23. ENGINE FUEL PRESSURE PUMP (REF)
24. FUEL-AIR HEAT EXCHANGER (REF)
25. TEMPERATURE GAGE (REF)
26. 46 MICRON FILTER (REF)
27. FUEL CONTROL (REF)
28. BYPASS PRESSURE REFERENCE LINE (REF)
29. FUEL-OIL HEAT EXCHANGER (ENGINE OIL) (REF)
30. FUEL FLOW TRANSMITTER (REF)
31. FUEL-OIL HEAT EXCHANGER (CONSTANT SPEED DRIVE UNIT OIL) (REF)
32. PRESSURIZATION AND DUMP VALVE (REF)
33. REGULATOR—FUEL TANK PRESSURE (2)
34. AMBIENT PRESSURE SENSING LINE (2)
35. REFUEL SHUTOFF VALVE (2)
36. PRESSURE REFUELING ADAPTER
37. PRESSURE REFUELING—DEFUELING ADAPTER
38. GRAVITY REFUELING ADAPTER (2)
39. GROUND JACK (4)
40. FUEL TRANSFER PUMP (4)
41. JETTISON SHUTOFF VALVE (2)
42. LIQUID SIGHT GAGE (DRIP STICK) (5)

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Fuel System Schematic
Figure 10-1 (Sheet 2 of 2)

Replenishment Tanks

Each fuel tank is composed of two separate sections, called the replenishment tank and the main tank. The replenishment tanks for fuel tank No. 1 and fuel tank No. 4 are located outboard of the No. 1 and No. 4 main tanks. Fuel flows from the two outboard replenishment tanks to the main tanks by gravity flow through check valves in the tank bulkheads.

The replenishment tanks for fuel tank No. 2 and fuel tank No. 3 are located forward of the No. 2 and No. 3 main tanks. Fuel is pumped from the inboard replenishment tanks into the main tank wells by transfer pumps as well as by gravity flow into the main tanks through check valves in the center spar.

Main Tank Wells

A "well" section, with an approximate capacity of 135 gallons, is located at the inboard aft corner low point of each main tank. Transfer pumps and check valves maintain the highest possible fuel level in the wells. Each well contains two fuel boost pumps which supply fuel to the engine fuel pumps.

Five separate subsystems comprise the main fuel system: Refueling-Defueling, Engine Fuel Supply, Fuel Jettison, Fuel Tank Vent and Fuel Quantity Gaging.

PRESSURE REFUELING SYSTEM

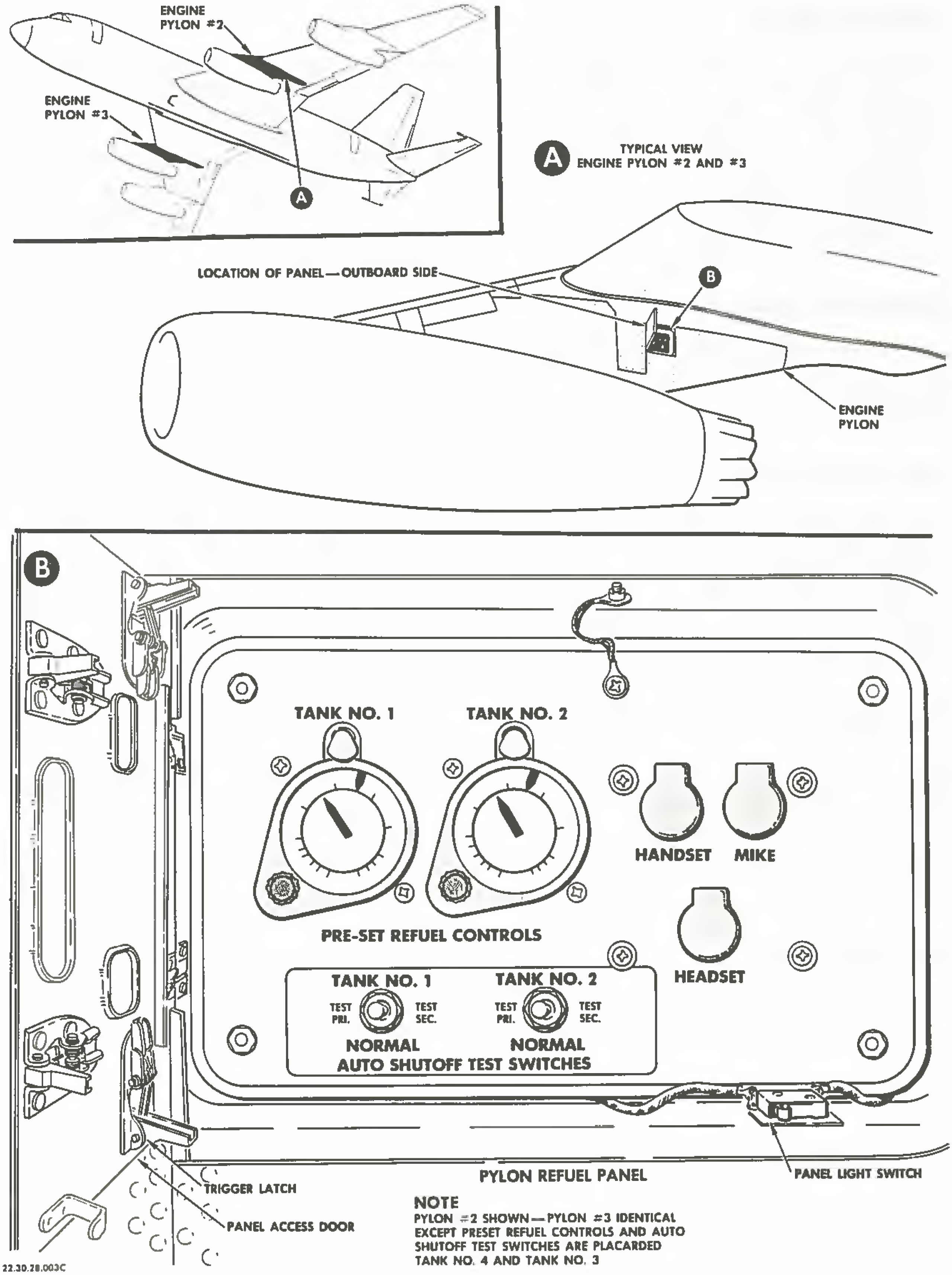
Pressure refueling is the normal method of refueling. Pressure refueling adapters are located in the lower surfaces of the wings outboard of the No. 2 and No. 3 engine pylons. Pressure hoses connect to the individual adapters for pressure refueling and tanks may be filled one at a time or simultaneously. The total intake is 1200 gpm, based on a 300 gpm flow with 50 psig fuel pressure at each of the four adapters.

CAUTION: MAXIMUM PRESSURE AT THE CONNECTORS SHOULD NEVER EXCEED 60 PSIG.

Two refueling control panels are provided, one in each inboard engine pylon With the airplane resting in a level attitude, the desired amount of fuel in pounds per tank is selected by presetting the quantity control knob located on the pylon fuel control panel (Figure 10-2).

GRAVITY REFUELING SYSTEM

Three-inch openings located in each of the four replenishment tanks, and accessible from the upper wing surfaces, are used for gravity-flow refueling. The fillers are capable of receiving fuel from two-inch diameter nozzles at the rate of 200 gpm. As fuel enters the replenishment tanks, the fuel level is equalized in the main tanks and tank wells through check valves in the separating bulkheads.



Pylon Refuel Control Panel
Figure 10-2

DEFUELING SYSTEM

Defueling at the rate of 50 gpm is accomplished through the inboard pressure fueling adapters by suction from the tanker. The defueling operation is controlled from the flight engineer's fuel control panel. Defueling of any or all four of the fuel tank systems may be accomplished through either of the inboard adapters by using the crossfeed valves. It is not necessary to close the corresponding emergency fuel shutoff valves in the pylons when defueling, since the positive displacement engine-driven fuel pumps will stop air leakage. Fuel is transferred from the desired tank through the defueling valve and into the fuel tanker or suitable receptacle. (see Figure 10-3).

ENGINE FUEL SUPPLY SYSTEM

Fuel from replenishment and main tanks is transferred by bulkhead check valves and transfer pumps to the main tank wells. From the main tank wells, dual booster pumps supply fuel to each of the four engine-driven fuel pumps (Figure 10-4).

Fuel Transfer Pumps

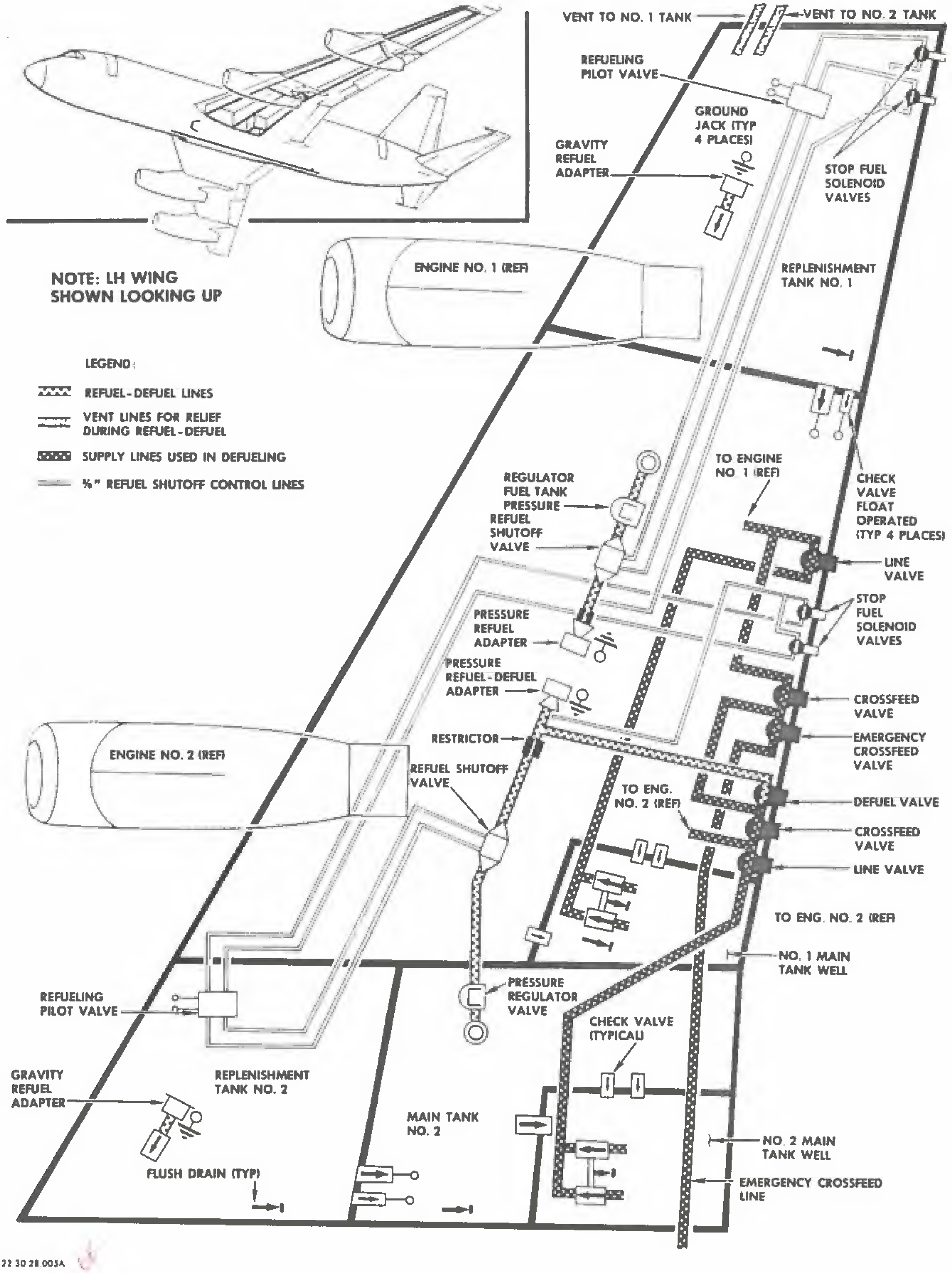
Two fuel transfer pumps are provided for each fuel tank. For the No. 1 and No. 4 tank systems (outboard), both pumps are located in the main tank; one in the aft outboard corner and one in the forward inboard corner. For the No. 2 and No. 3 tank systems (inboard), one pump is located in the aft inboard corner of the replenishment tank and one pump in the forward inboard corner of the main tank. Fuel transfer pumps are used to transfer fuel from their respective areas into the main tank wells.

Replenishment Tank Fuel Level Indication

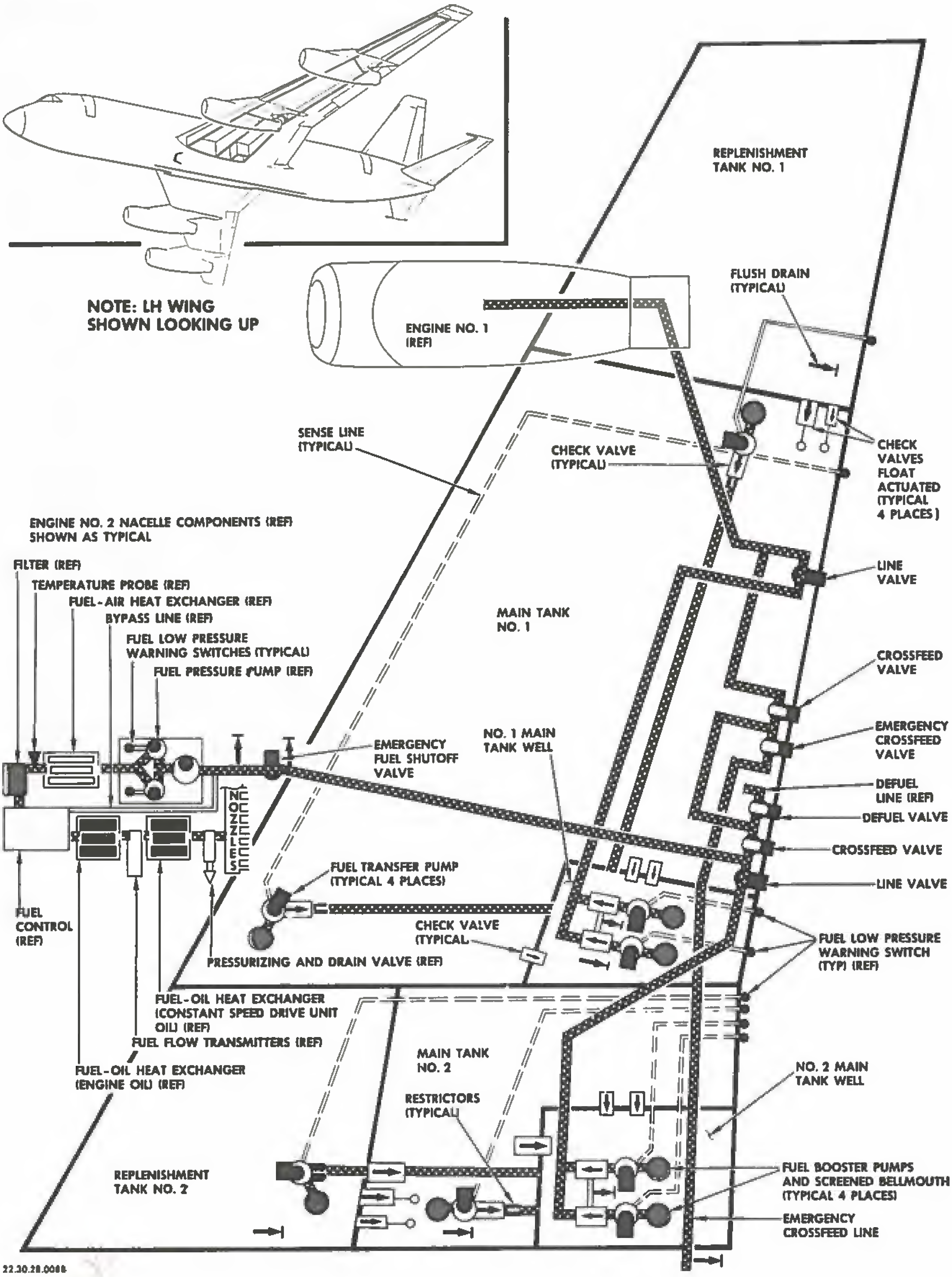
Low fuel level in the inboard replenishment tanks is indicated by the illumination of the pump low pressure warning light for the transfer pump of each inboard system. Illumination of the light indicates that the pump discharge pressure is low, either from lack of fuel or pump malfunction. Float switches in series with the pump low pressure warning light switches of all main tank transfer pumps prevent the corresponding warning light from illuminating unless the pressure loss is due to pump malfunction.

Fuel Boost Pumps

Each engine is normally supplied with fuel from two boost pumps located in each main tank well. The pumps maintain between 15 psia (minimum) and 50 psig (maximum) fuel pressure at each engine fuel pump inlet during all airplane operating conditions. Both, or either pump alone, are capable of supplying a maximum fuel flow rate of 125 percent of takeoff requirements (standard day fuel consumption). Any pair of boost pumps is capable of supplying 100 percent maximum takeoff fuel



Refueling-Defueling
System Schematic
Figure 10-3



Fuel Supply System Schematic
Figure 10-4

flow to its own engine and to one engine in the opposite wing. Any pump pair is capable of supplying 100 percent of maximum cruise fuel flow to all four engines.

Emergency Fuel Shutoff Valves

An emergency fuel shutoff valve is installed above and adjacent to each engine pod firewall. The valves are operated in connection with the fire protection system. Their primary function is to close off fuel to the engines in event of a fire. Operating the FIRE PULL "T" handle of the fire protection system causes the corresponding emergency fuel shutoff valve to close.

Engine Fuel and Control System Block Diagram

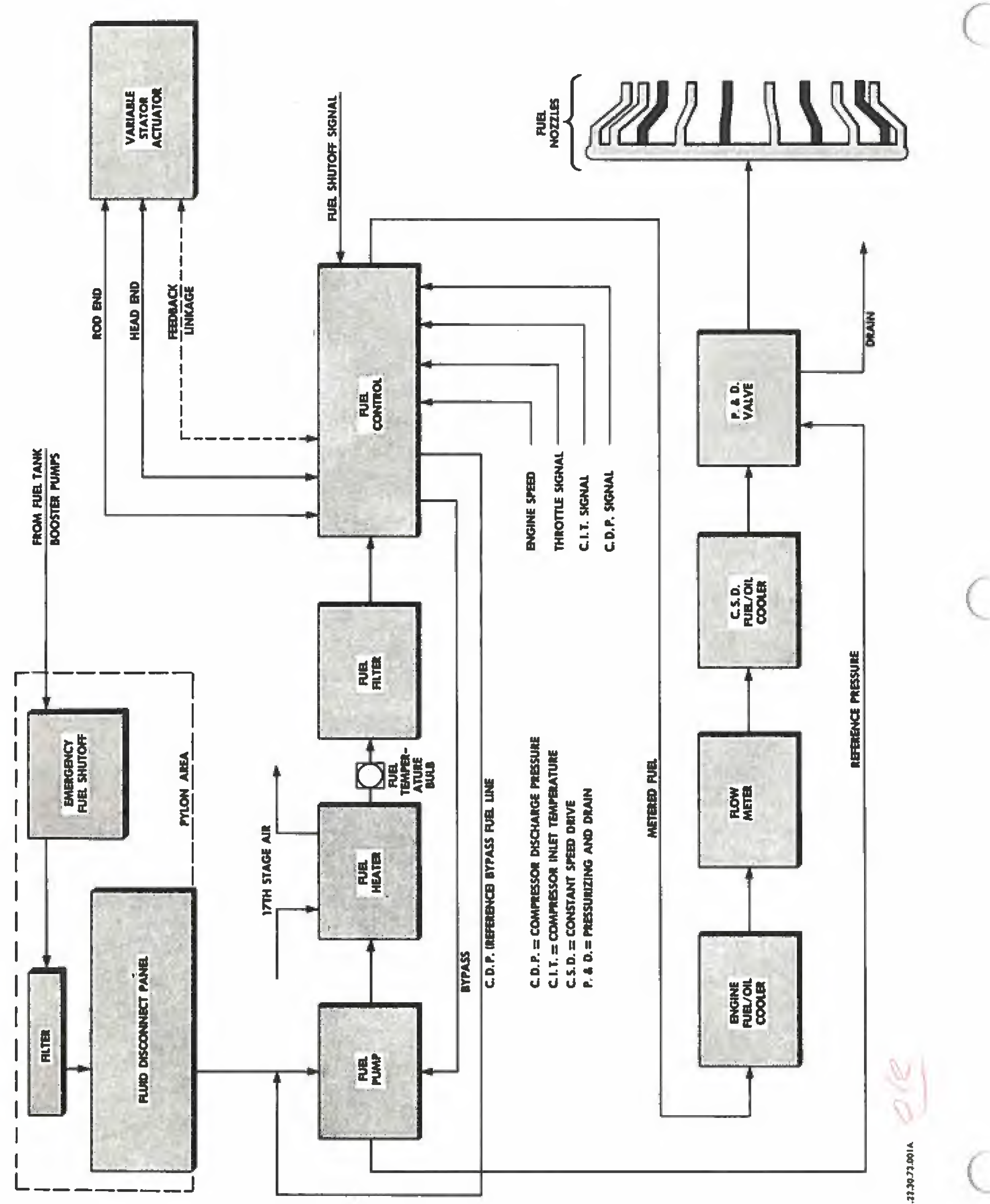
Figure 10-5, Engine Fuel Supply Block Diagram, provides a ready reference as to engine fuel flow from the firewall emergency fuel shutoff valve to the fuel nozzles.

Engine Fuel Pump

A primary boost element in the engine fuel pump increases the fuel tank boost pump pressure by 15 to 45 psi. The fuel flows from the primary boost element to two high pressure elements, increasing the fuel pressure 100 to 850 psig depending on engine speed and flight conditions. The pump discharges the fuel to the fuel heater. Fuel flows through the heater and into the fuel filter. From the filter the fuel goes into the fuel control, to the engine fuel-oil cooler, CSD fuel-oil cooler, the pressurization and drain valve, and into the fuel nozzles. The fuel pump always delivers more fuel to the control than is required to sustain the engine under any condition. The control uses some of the excess fuel to operate internal servo mechanisms and to position and cool the engine variable stator actuators. The remainder is returned to the inlet side of the high pressure elements of the fuel pump. In case of failure of both fuel tank boost pumps in any one tank, the engine-driven fuel pump supplies sufficient fuel to the engine to maintain all performance requirements up to 30,000 feet. Fuel temperature at the engine must be approximately 43 degrees C (110 degrees F).

Engine Fuel Heater

The engine fuel heater is a counter-flow air-to-liquid type heat exchanger. Fuel enters through the fuel heater inlet and flows through the fuel tubes. Seventeenth stage compressor air at approximately 63 degrees C (145 degrees F) flows around the fuel tubes warming the fuel. The fuel flows past a temperature sensor and is discharged to the fuel filter. Expansion or contraction of the thermostatic element in the fuel temperature sensor opens or closes the fuel heater compressor air shutoff valve. The valve closes at fuel discharge temperatures above 7 degrees C (45 degrees F) and opens at fuel discharge temperatures below 0 degrees C (32 degrees F).



Engine Fuel Supply Block Diagram
Figure 10-5

Engine Fuel Filter

The fuel filter, located between the fuel heater and the fuel control, removes solid contaminants from the fuel. Fuel enters the filter and surrounds the filter screen, passes through the screen into an inner chamber, and flows out the discharge port. If the filter should become clogged, fuel will bypass through the relief valve which opens when the pressure differential across the filter is 20 to 24 psi.

Engine Fuel Control

The engine fuel control is a hydro-mechanical system. High turbojet engine performance calls for rapid engine acceleration, but a conventional seventeen stage fixed compressor would tend to stall during throttle bursts. The problem is caused by critical combinations of air, temperature and engine speed which create air turbulence and consequently restrict the flow of air through the compressor during acceleration. When this condition develops, the turbojet engine usually stalls or fails in rapid acceleration. This difficulty is overcome in the CJ805-3 engine by making the twenty inlet guide vanes and the first six stages of compressor stator blades variable. As some point is reached during acceleration (depending on compressor inlet temperature), the variable stators rotate slightly to maintain an angle with the inlet air that encourages a smooth airflow and at the same time discourages air turbulence. The fuel control unit is instrumental in positioning the inlet guide vanes and the stator blades. Thus the fuel control system can be considered made up of two subsystems: the variable stator subsystem which is concerned with the continuous and automatic adjustment of the airflow path, and the fuel subsystem which is concerned with fuel metering and delivery to the combustion chambers.

Specific discussion of the complex engine fuel control system can be found in the appropriate engine manual. However, briefly, the fuel control system includes components which sense compressor inlet temperature (CIT), compressor discharge pressure (CDP), engine rpm, fuel temperature, specific density of the fuel being used, and power lever signals. During flight, as the ambient temperatures and pressures change, the fuel control system will automatically vary both fuel flow and compressor stator angle in order to operate the engine efficiently at the rpm selected by the pilot through setting of the power control lever.

Fuel Shutoff Levers

The fuel shutoff lever actuates a fuel shutoff valve in the discharge portion of the fuel control unit. The valve has two positions; RUN (open position), and OFF (closed position). Metered fuel from the fuel control flows through the fuel shutoff valve to the engine fuel-oil cooler. The engine must never be started or operated with the fuel shutoff lever in an intermediate position. Use as a fuel throttling means will damage the fuel control unit.

Engine Fuel-Oil Cooler

The engine fuel-oil cooler is a liquid-to-liquid heat exchanger using engine fuel as the coolant. (See Section 11, OIL SYSTEM for detailed information). From the engine fuel-oil cooler the fuel is directed through the fuel flow transmitter to the constant speed drive (CSD) fuel-oil cooler.

Fuel Flow Transmitter

The fuel flow transmitter is a propeller-driven synchro motor that is actuated by varying fuel flow against the propeller. The transmitter signal is displayed on the pilots' engine instrument panel in pounds per hour of fuel flow.

Constant Speed Drive Fuel-Oil Cooler

The constant speed drive fuel-oil cooler is a liquid-to-liquid type heat exchanger using engine fuel as the coolant. From the constant speed drive fuel-oil cooler, metered fuel is routed to the pressurizing and drain valve.

Fuel Pressurizing and Drain Valve

Metered fuel flows to the inlet of the fuel pressurizing and drain valve. When the inlet pressure rises to 80 to 100 psi above the reference pressure plus the spring force that is applied to the aft side of the pressurizing piston, the drain piston closes the drain port. The pressure also moves the pressurizing piston away from the drain piston, allowing fuel to flow to the engine. When the pressure differential across the piston drops below 80 psi, fuel flow to the engine is cut off and the drain port is opened to drain the fuel manifold.

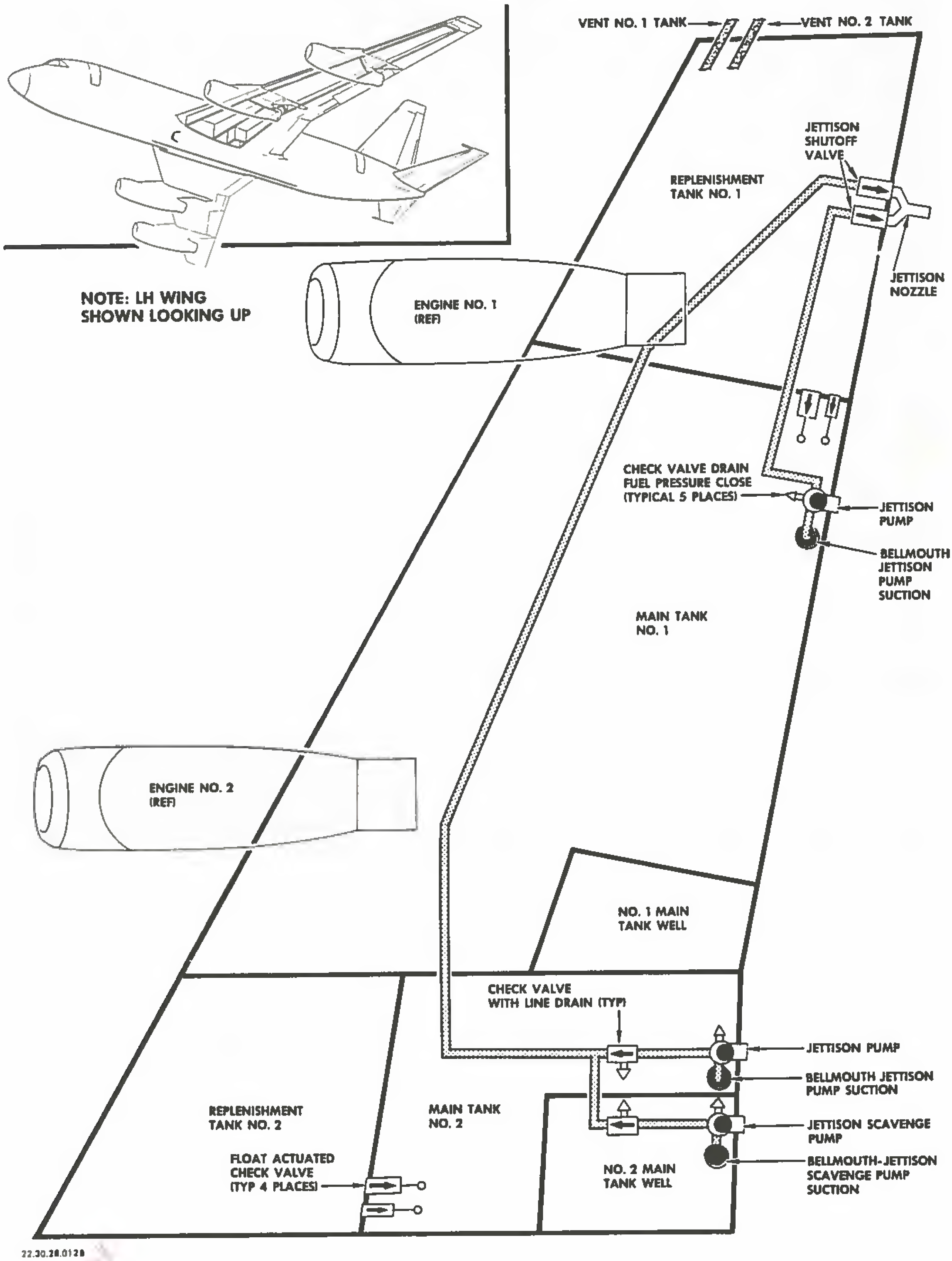
The total drainage after engine shutdown or a wet prestart is 450 to 550 cc. No leakage should occur during other phases of engine operation.

Engine Fuel Nozzles

Metered fuel from the pressurizing and drain valve flows into the ten fuel nozzles which spray atomized fuel into the combustion chambers.

FUEL JETTISON SYSTEM

The fuel jettison system is designed for fuel jettisoning and for fuel scavenging. The fuel jettison system permits in-flight reduction of the airplane weight to the maximum permissible landing weight. The scavenge portion of the system completely empties the inboard tanks, adjacent to the cabin area, in the event that an emergency wheels up landing is anticipated. A schematic of the system is shown in Figure 10-6. The jettison/scavenge system is a fixed installation useable at any airplane configuration of flaps or landing gear and at speeds up to V_{NO}/M_{NO}.



Fuel Jettison System Schematic
Figure 10-6

A hydraulically operated fuel jettison pump is provided in each of the four main fuel tanks. The jettison pumps are controlled from switches located on the fuel control panel. Moving the switches to the OPEN position energizes solenoid valves which port No. 1 hydraulic system fluid to the jettison pump drive motors. Each fuel tank jettison system is equipped with inlet bellmouth fittings positioned to assure a minimum of 8810 pounds total fuel remaining in the tanks after jettisoning. The jettison system is capable of dumping overboard a minimum of 49,800 pounds of fuel in 27 minutes, or an average of 1845 pounds per minute.

Scavenge System

A hydraulically operated fuel scavenge pump is located in each inboard main tank well. The two scavenge pumps are controlled by switches located on the flight engineer's fuel control panel. These switches actuate solenoid valves which port No. 2 hydraulic system fluid to the scavenge pump motors. The scavenge pumps obtain fuel from the bottom of the wells through the downturned bellmouth inlets and scavenge the entire inboard system (tanks No. 2 and No. 3) of each wing. This leaves only the fuel remaining below the bellmouths in the outboard systems (tanks No. 1 and No. 4) for operation during emergency wheels up landing procedures.

Scavenge/Jettison Systems Hydraulic Power Source

The fuel jettison system is operated by hydraulic power from the No. 1 hydraulic system. The fuel scavenge system is operated by hydraulic power from the No. 2 hydraulic system.

Fuel Jettison Nozzles

A fuel jettison nozzle is located in the trailing edge of each wing near the wing tip. Each nozzle is fed by a fuel jettison line. The scavenge fuel line is teed into the jettison line for the inboard tank. The nozzles supply a proper exit stream pattern and establish a minimum exit velocity of 30 feet per second to prevent reverse fuel flow in the wing boundary layer and to retard flame propagation. The nozzles are permanently fixed and streamlined aft of the trailing edges.

FUEL TANK VENT SYSTEM

Three vent valves are used in each tank vent system. One float-type valve is located in each main tank and in each replenishment tank, and one combination float-type and pressure relief valve is located in each replenishment tank. The tank venting systems connect with the ram-vent scoop in the lower surface of each wing tip.

Ram-Vent Scoop

The ram-vent scoop is fitted flush with the lower surface of each wing tip and serves two purposes. Its purpose is to vent vapor pressure from the tanks; in addition, the scoop has a plenum chamber which scoops air during flight to the extent that approximately two to three pounds of air pressure above ambient is maintained in the tank systems.

FUEL CONTROL PANEL

All functions of the fuel tank systems except refueling are controlled from the flight engineer's fuel control panel (see Figure 10-7). The fuel control panel contains fuel quantity indicators, valve position indicators, fuel pressure warning lights, boost and transfer pump switches, and normal feed, crossfeed, defuel, scavenge and jettison valve switches. A single fuel temperature indicator is also installed on the panel.

The arrangement of controls, indicators and warning lights permits an immediate interpretation of the fuel feed condition being used, as well as the immediate selection of any other configuration desired. A schematic diagram of the fuel line circuit is superimposed on the panel and is shown on an internally illuminated diagram. The fuel valve switch knobs are placed across the circuit to represent the corresponding valve location in the actual fuel system. The switch knobs incorporate an engraved solid line across their circumference. By aligning the line on a knob with the fuel circuit line, the corresponding valve is opened; when the knob reference line is set at right angles to the fuel line on the panel, the valve is closed.

Fuel Quantity Gaging System

The fuel quantity gaging system indicates fuel quantity in pounds for all four of the tank systems. A fuel quantity gage for each tank is located on the flight engineer's fuel control panel. A spring-loaded three-way switch in the center of the fuel control panel selects the mode of operation of the four fuel quantity gages. Held in the MAIN position, all gages read fuel quantity in the main tanks. Held in the REPLENISH position, all gages read fuel quantity in the replenishment tanks. In the normal spring-loaded TOTAL position, each gage reads total fuel quantity in the main plus replenishment tanks for its respective tank system. The totalizer fuel indicator on the pilots' center instrument panel displays total fuel quantity on board, or total fuel in main or replenishment tanks when these tanks are selected by the flight engineer. Repeater indicator gages in the pylon refuel control panels on the inboard pylons also indicate fuel quantity. There is one gage for each left tank system in the left pylon refuel panel and one gage for each right tank system in the right pylon refuel panel. Drip sticks, ten per airplane, are provided in various wing access doors.

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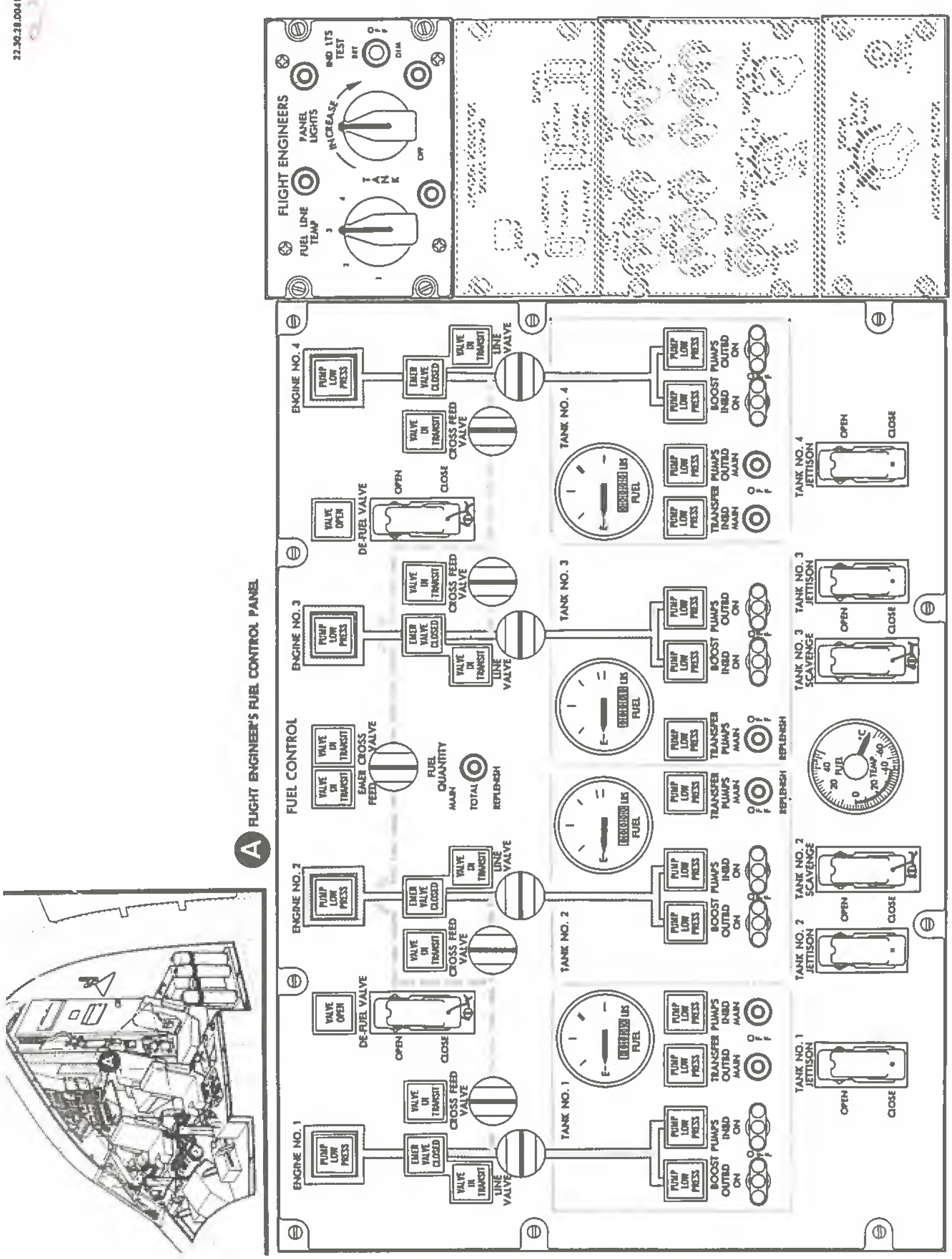


Flight Engineer's Fuel Control Panel
Figure 10-7

Pump Low Pressure Warning Lights

An amber PUMP LOW PRESS indicator light is provided for each of the six fuel transfer pumps and eight red PUMP LOW PRESS warning lights for the fuel boost pumps. The No. 2 main transfer pump and the No. 2 replenish tank transfer pump have the same warning light. The No. 3 main transfer pump and the No. 3 replenish tank transfer pump have the same warning light.

The eight boost pump warning lights illuminate when the fuel pressure drops due to lack of fuel or malfunction of the pump.

Except for No. 2 and No. 3 replenishment tank transfer pumps, all transfer pump lights illuminate only when the pump malfunctions. Float switches prevent their illumination for low fuel conditions. The No. 2 and No. 3 replenishment tank transfer pump lights will illuminate for low fuel conditions as well as pump malfunctioning.

Engine Fuel Pump Warning Lights

One red PUMP LOW PRESS warning light is provided for each engine driven fuel pump. Illumination indicates either low fuel condition or pump malfunction.

Emergency Fuel Shutoff Indicator Lights

One green EMER VALVE CLOSED indicator light is provided for each engine. Illumination of this light indicates the emergency fuel shutoff valve, located at each engine firewall, has been closed by operation of a FIRE-PULL "T" handle.

Valve In Transit Indicator Lights

Each of the ten fuel crossfeed and line valves is provided with a blue VALVE IN TRANSIT indicator light. The valve in transit light illuminates only during the time the valve is in motion and extinguishes when movement is completed. Two indicator lights are provided for the emergency crossfeed valve switch since actuation of this switch controls a valve in each wing.

De-Fuel Valve Indicator Lights

A red VALVE OPEN warning light is provided for each de-fuel valve. Each light illuminates when the associated de-fuel valve is open.

De-Fuel Valve Switches

Two de-fuel valve guarded switches are provided and are safetied in the CLOSE position for normal operation. Moving the switches to the OPEN position opens the two de-fuel valves.

Line Valve Switches

Turning the line valve switches to align the knob marker with the circuit diagram fuel line opens the line valve. Turning the knob marker to a position across the circuit diagram fuel line closes the line valve.

Crossfeed Valve Switches

Turning the crossfeed valve switches to align the knob marker with the circuit diagram fuel line opens the crossfeed valve. Turning the knob marker to a position across the circuit diagram fuel line closes the crossfeed valve.

Boost Pump Control Switches

The boost pumps are started by eight boost pump control switches on the fuel control panel. Each switch incorporates an OFF and ON position.

Tank No. 1 and Tank No. 4 Transfer Pump Switches

Two transfer pump switches are provided for tank No. 1 and two transfer pump switches for tank No. 4. Moving a switch from OFF to the ON position starts the corresponding transfer pumps.

Tank No. 2 and Tank No. 3 Transfer Pump Switches

Tank No. 2 and tank No. 3 are each provided with a single three-position transfer pump switch. The center position is the OFF position. Movement to the MAIN position operates the transfer pump located in the main tank. Movement to REPLENISH position operates the transfer pump located in the replenishment tank.

Jettison Switches

A tank jettison guarded switch is provided for each tank and each switch is safetied in the CLOSE position. Movement of the switch to the OPEN position opens a solenoid valve and port No. 1 hydraulic power system fluid to the appropriate jettison pump.

Scavenge Switches

Two scavenge guarded switches are provided, one for tank No. 2 and one for tank No. 3. Both switches are safetied in the CLOSE position. Movement of the switches to the OPEN position opens a solenoid valve and port No. 2 hydraulic power system fluid to the appropriate scavenge pump.

Fuel Temperature Indicator

The fuel temperature gage on the fuel control panel indicates fuel temperature in degrees Centigrade as read by any one of five temperature sensitive elements. One element is located in the No. 1 tank well and the other four are located, one to each engine, in the fuel stream near the engine-driven fuel pump. A five position selector switch is provided on the right side of the fuel control panel.

PYLON REFUELING CONTROL PANEL

The refueling control panel located on the aft outboard side of each inboard pylon contains fuel quantity gages, preset quantity selectors, and precheck shutoff controls. Jacks for microphone, handsets and headsets are installed for direct communication with the flight compartment only during defueling operations. During pressure refueling operations, personnel are not required in the flight compartment and the entire operation is conducted from the refueling control panel. It is impossible to replace the cover on the refueling control panel unless all controls, including removal of defueling communication equipment, are in proper position for flight (see Figure 10-2).

Fuel Quantity Indicators

Two fuel quantity gages are provided for each refueling control panel, one gage for the outboard tank system and one gage for the inboard tank system. The inner pointer of each gage displays the total quantity of fuel in its associated tank. The outboard pointer of each gage is capable of being set to the desired refueling quantity by a knob at the lower left corner of each indicator. When the two pointers coincide, the refueling flow will be stopped.

CAUTION: WHEN THE PRESET FUEL QUANTITY HAS BEEN DELIVERED TO THE FUEL TANKS AND AUTOMATIC SHUTOFF HAS BEEN ACCOMPLISHED, THE FUEL SUPPLY SOURCE PRESSURE SHOULD BE RELEASED IMMEDIATELY.

Precheck Switches

A three-position precheck switch is provided to test the automatic refueling system for each tank. Moving the switch to the STOP "P" position will open the primary solenoid valve and port pressure fuel to the primary side of the dual pilot-level control valve, resulting in shutoff of fuel flow. Moving the switch to the STOP "S" position will open the secondary solenoid valve and port pressure fuel to the secondary side of the dual pilot-level control valve, resulting in shutoff of fuel flow. Setting the switch on AUTO position allows the "stop" signal, determined by setting of the refuel-quantity pointer, to be applied to the primary and secondary solenoid valves after the selected amount of fuel has been delivered to the tank.

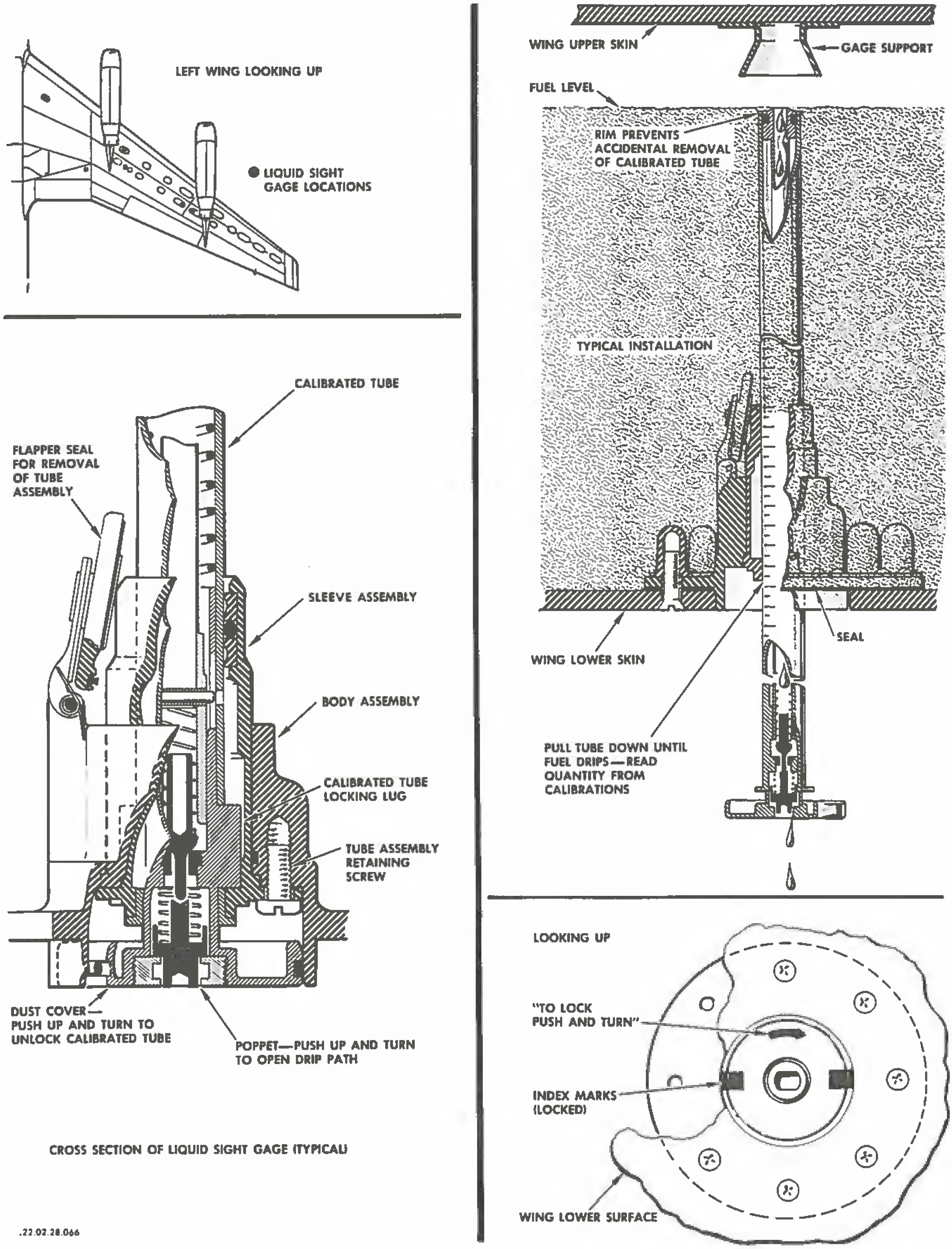
LIQUID SIGHT GAGES

A total of 10 liquid-sight gages are provided. The gages are installed in housings attached to the bottom of the tanks and extend up through the tanks, fitting into gage supports attached to the top of the tanks (see Figure 10-8). Each liquid sight gage consists of a calibrated tube, or drip stick, which is free to slide up and down within a sleeve attached to the gage housing. The bottom of the tube is attached to a butt plate which fits into a recess in the under surface of the wing.

The gage is unlocked by pushing the butt plate upward and turning it to the left with a special tool. The tube, with the tool attached, is lowered until fuel drips from the tube indicating that the top of the tube is below the fuel level in the tank. The tube is then raised until the dripping ceases and the quantity of fuel in the tank is read in pounds at the highest visible calibration on the tube. A trap attached to the tool catches the minor fuel leakage when the tube is lowered below the surface of the fuel.

The gage is closed by pushing the butt plate up into the recess in the wing surface and turning it to the right until the index marks on the butt plate and the collar are aligned. Because of the height of the sleeve and housing; the fuel level cannot be measured below 2.9 inches above the bottom of the tank.

The tube and sleeve assembly can be removed for replacement without draining the tank.



Liquid Sight Gage
Figure 10-8

FUEL SYSTEM ELECTRICAL SOURCES

For detailed circuit information, consult the WIRING DIAGRAM MANUAL.

The control relay coils for each boost and transfer pump are energized through the individual switches from the following bus:

1. From the 115-volt, 400 cps, nonessential ac bus:
   No. 1 tank outboard boost pump
   No. 2 tank outboard boost pump
   No. 3 tank outboard boost pump
   No. 4 tank outboard boost pump
   No. 1 main tank inboard transfer pump
   No. 2 main tank transfer pump
   No. 3 main tank transfer pump
   No. 4 main tank inboard transfer pump

2. From the 115-volt, 400 cps, essential ac bus:
   No. 1 tank inboard boost pump
   No. 2 tank inboard boost pump
   No. 3 tank inboard boost pump
   No. 4 tank inboard boost pump
   No. 1 main tank outboard transfer pump
   No. 2 replenish tank transfer pump
   No. 3 replenish tank transfer pump
   No. 4 main tank outboard transfer pump

Boost and Transfer Pump Operating Power Sources

All boost and transfer pumps are operated by 200/115-volt, 3-phase, 400 cps ac electrical power. Each phase to each pump is individually protected by circuit breakers.

Defueling Valve Power Source

Both defueling valves are operated by 28-volt dc. If required, the airplane emergency battery dc power can be used.

Fuel Tank Shutoff Valve Power Source

All fuel tank shutoff valves are operated by 28-volt dc. If required, the emergency battery dc power can be used.

Crossfeed Valve Power Source

All crossfeed valves are operated by 28-volt dc. If required, the emergency battery power can be used.

Emergency Fuel Shutoff Valve Power Source

All emergency fuel shutoff valves, operated by the fire-pull "T" handles, are operated from the 28-volt dc emergency bus. If required, the emergency battery power can be used.

Fuel Jettison System Power Source

All fuel jettison system pump solenoid valves are operated by 28-volt dc. Actual operation of the jettison pumps is by hydraulic fluid ported to the pumps by opening of the solenoid valves.

Fuel Quantity Indicator System Power Source

Power for the fuel quantity selector switch system is 28-volt dc. Power for operation of the probe-indicating systems is 115-volt, 400 cps.

Fuel Pump Low Pressure Warning Light Power Source

The engine fuel pump low pressure warning lights operate from 28-volt dc.

The fuel tank booster and transfer pump low pressure warning lights operate from 28-volt dc.

Fuel Temperature Indicator Power Source

The fuel temperature indicator operates from 28-volt dc.

Pylon Refueling System Power Source

The pylon refueling system operates from 28-volt dc.

Fuel Flow Indicator Power Source

The fuel flow indicators are operated by 115-volt ac, 400 cps.

Section 11

OIL SYSTEM

TABLE OF CONTENTS

OIL SYSTEM

GENERAL OIL SYSTEM

The oil system is composed of three subsystems; the engine system, the constant speed drive (CSD) hydraulic system and the thrust reverser actuation system. The engine oil system is used to cool and lubricate the bearings and gears. With respect to the constant speed drive and the thrust reversers, the oil is utilized as a hydraulic fluid and lubricant for the CSD power drive and for hydraulic actuation purposes. (see Figure 11-1).

Oil System Reservoir

The oil system reservoir is mounted on the upper forward right side of the engine (see Figure 11-2). The reservoir is of single unit construction with an internal vertical bulkhead that divides the interior into two separate reservoir sections. The two sections are connected by an internal pressure equalizing line in the uppermost portion of the expansion spaces. The larger and aft section of the reservoir holds 4.15 gallons of oil for the engine system. The forward section of the reservoir holds 1.72 gallons of oil for the constant speed drive and thrust reverser systems. There is no flow connection between the oil in the two reservoir sections or between the systems they supply.

Oil System Replenishment

Each section of the oil reservoir is provided with a gravity flow filler port at the top of the reservoir. A calibrated dipstick is provided as an integral portion of each gravity fill cap. Accidental overflow of oil is removed by a scupper drain opening between the two caps with the drain exit at the bottom of the reservoir.

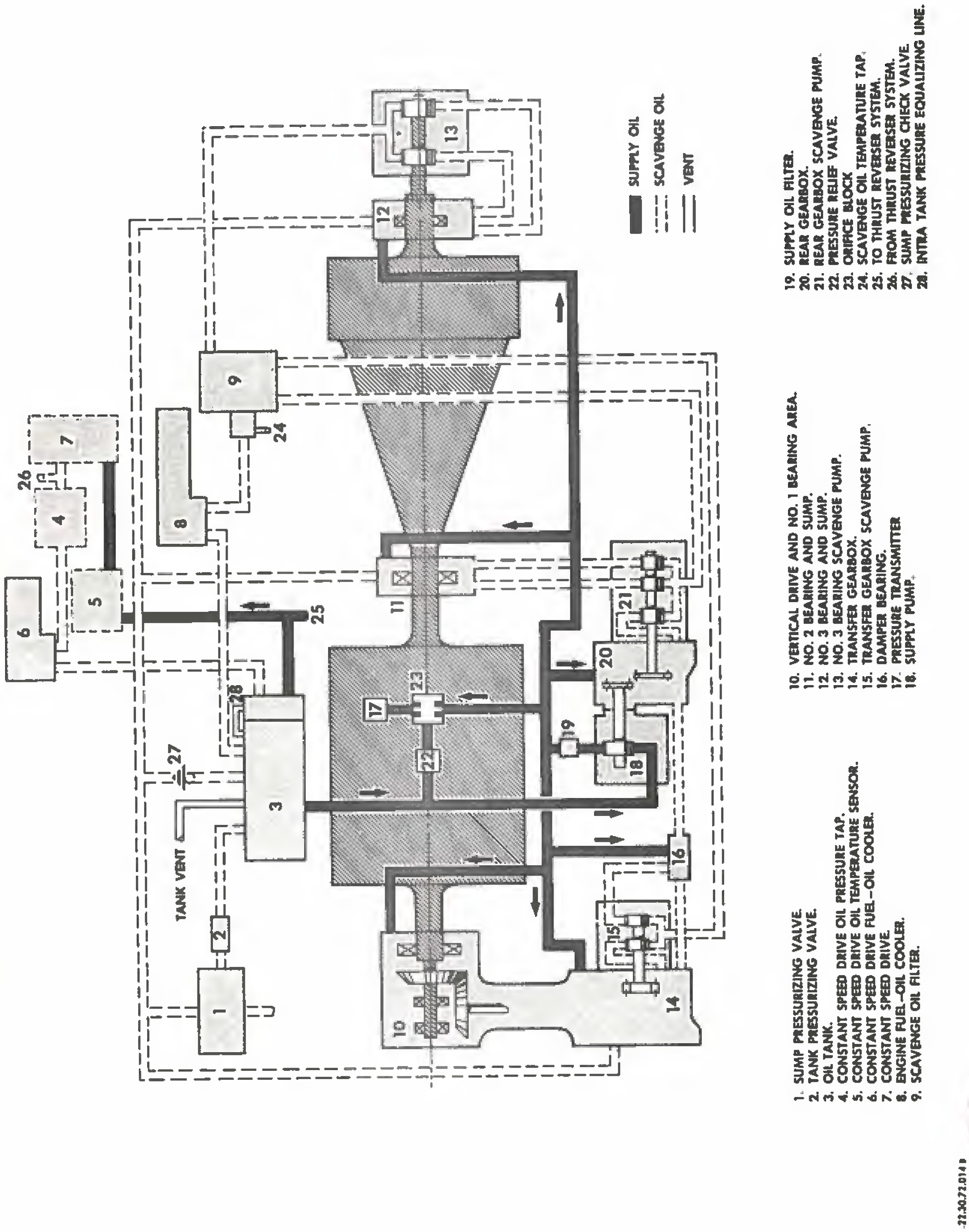
When the engine oil section of the reservoir is filled to 4.15 gallon capacity, 1.26 gallons expansion space is provided. The CSD section, when filled to 1.72 gallon capacity, provides 0.50 gallons of expansion space.

Oil System Consumption Rate

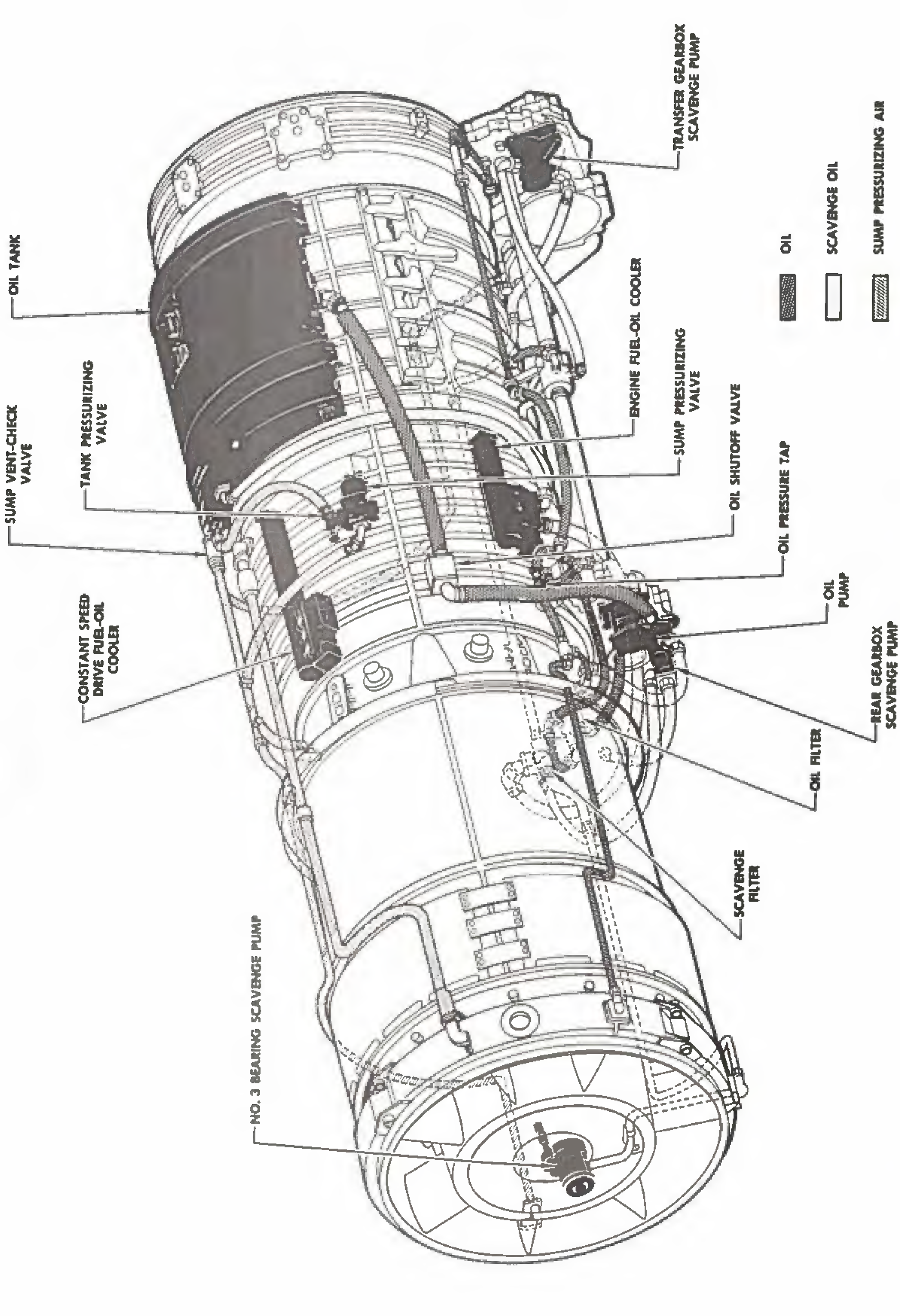
With the oil reservoir filled to capacity, the engine has a flight capability of 13.6 hours. At the end of that period approximately 1.15 gallons of oil will be left in the engine section. With this quantity in the reservoir, the engine can be inclined 30 degrees up or down, or rolled to the right or left 20 degrees without uncovering the oil outlets.

ENGINE OIL SYSTEM

Oil from the engine section of the oil reservoir flows to the engine oil pump. The engine-driven positive displacement pump increases the oil pressure to 12 to 65 psig, depending on engine speed, and discharges the oil through a check



Oil System Schematic Diagram
Figure 11-1



Oil System Components
Figure 11-2

valve to the engine oil filter. The filtered oil is then divided into flows supplying the front gearbox and the No. 1 bearing, the transfer and rear gear boxes, the horizontal drive shaft damper bearing, and the No. 2 and No. 3 bearings. A pressure relief valve bypasses oil from the downstream side of the oil filter to the inlet side of the oil pump in case of filter clogging. In this way, the maximum pressure delivered to the oil pressure indicating system is 200 psig. Actually, insofar as pump performance is concerned, during cold weather starting, engine oil pressure may range as high as 500 to 600 psig.

Engine Oil Scavenge System

The engine oil in the bearing sumps, damper bearing and accessory cases is scavenged, filtered, cooled, de-aerated and returned to the engine section of the oil reservoir by means of the three scavenge system oil pumps. The scavenge system pumping capacity is approximately 2.5 times the actual amount of oil supplied to any one area. The scavenge oil filter contains a magnetic chip detector which attracts any metal chips in the filter. An accumulation of chips will complete a metallic path which can be detected by use of an external continuity check. Thus the presence of metallic contamination can be determined without removing and inspecting the filter.

Engine Fuel-Oil Cooler

The engine fuel-oil cooler is a liquid-to-liquid heat exchanger which uses engine fuel as a coolant. When the scavenge oil temperature is below 38 degrees C (100 degrees F), a bypass valve is opened allowing a portion of the oil to flow directly to the cooler outlet without passing through the cooler. As the temperature increases, the valve closes, forcing all the oil to flow through the cooler core. A fuel temperature sensor opens the bypass valve when the fuel temperature reaches 116 degrees C (241 degrees F). Since certain engine operating conditions may demand greater fuel flow through the cooler than is possible through the normal core system, a bypass valve is provided that will open and allow large volumes of fuel to bypass the cooler.

CONSTANT SPEED DRIVE (CSD) OIL SYSTEM

The CSD oil system provides lubrication, control system hydraulic power and transmission hydraulic power. The oil from the CSD section of the reservoir is increased in pressure to 140 to 150 psig by the CSD internal oil pump. From the pump, the pressurized oil is directed through a filter unit, past a magnetic chip detector and into the control, lubrication and actuating portions of the CSD unit. A temperature sensing probe projects into the CSD sump area. The thermal switch and probe assembly will illuminate a DRIVE MALFUNCTION light on the flight engineer's panel when the oil temperature reaches or exceeds 177 degrees C (350 degrees F).

CSD Scavenge Pump

The CSD scavenge pump scavenges oil from the CSD case sump and returns it to the CSD internal reservoir. When the internal reservoir is full, a relief

valve operating from 110 to 120 psi ports the excess oil to the CSD section of the main oil reservoir through the oil-to-air and oil-to-fuel coolers.

CSD Oil Coolers

CSD oil from the CSD internal reservoir is directed to the CSD oil-air cooler. A bypass valve controlled by oil temperature is provided on the oil-air cooler. From the oil-air cooler the oil flows to the CSD oil-fuel cooler. However, the amount of heat the fuel can absorb at this point in the fuel system flow approaches zero under certain engine operating conditions. A fuel temperature sensor in the oil-fuel cooler opens the oil bypass valve when the fuel temperature reaches 116 degrees C (241 degrees F) and ports the oil directly to the CSD section of the oil reservoir.

OIL SYSTEM PRESSURIZATION

The sump pressurizing system regulates the pressure of the air in the oil tank, the gearboxes and bearing sumps. These units are pressurized primarily to assure a positive pressure at the inlet of the pressure and scavenge pumps. However, the method used to obtain air for pressurization is based upon the premise that airflow is from the rear side of the seals into the sump area. The results of this flow is a control of oil consumption--since, with air continuously flowing into the bearing sumps, only a small amount of oil is lost through the seals. By maintaining sump and tank pressure at 3 to 7 psi above ambient pressure, a balanced system is provided.

The air used for pressurization is obtained from two sources; leakage of air across the air and oil seals into the sumps and/or ambient air. The source of the air depends largely upon the efficiency of the seals; if sufficient air bleeds across them to prevent cavitation of the pumps, no ambient air is required. However, if the seals are exceptionally tight and virtually no air bleeds across them, ambient air must be drawn into the system.

Altitude Sensing and Pressurization

All of the scavenge pumps on the engine are capable of pumping a greater quantity of oil than is present in the sumps--the deficit is made up of air. As the scavenge oil-air mixture passes through the de-aerator in the oil tank, the entrained air is separated from the oil. The air, plus any excess air that may enter the tank from the sump vent line, pressurizes the oil tank. The only route of escape for the air in the tank is through the tank and sump pressurizing valves. The overboard vent of the sump pressurizing valve is full open up to 20,000 feet and pressure in the tank and sumps is established at 3 to 4 psi above ambient pressure by the action of the tank pressurizing valve. At 20,000 feet, the sump pressurizing valve begins to regulate to increase pressure in the system. At 28,000 feet, the valve attains full regulation and, added to the action of the tank pressurizing valve, establishes pressure in the tank and sumps at 5 to 7 psi above ambient pressure.

OIL SYSTEM DESIGN LIMITS

Normal limits shown below define a range that is encountered during normal operations. Extreme limits define a range that may be encountered less than 5 percent of the operating time.

1. Lubrication oil temperature into the engine
   Normal..... 38 degrees C to 122 degrees C (100 degrees F to 250 degrees F)
   Extreme... -40 degrees C to 149 degrees C (-40 degrees F to 300 degrees F)

2. Lubrication oil temperature out of the engine
   Normal.... 66 degrees C to 149 degrees C (150 degrees F to 300 degrees F)
   Extreme... -40 degrees C to 177 degrees C (-40 degrees F to 350 degrees F)

3. Lubrication oil pressure (Indicated)
   Normal.... 12 psig minimum at IDLE rpm
   35 to 65 psig at normal rated thrust
   Extreme (Cold Start)...... 200 psi (gauge limit)

Oil System Flight Attitude Limits

The oil system will perform satisfactorily during continuous engine operation in the following attitudes:

1. Level fore and aft, 20 degree roll attitude in either direction.
2. Dive or climb angle of 0 to 30 degrees, 10 degree roll attitude in either direction.

THRUST REVERSER OIL SYSTEM

The engine thrust reverser system has its own oil pump and oil filter. Oil flows from the CSD section of the oil reservoir to the reverser oil system engine-driven pump which increases the oil pressure to 800 to 1300 psig at 100 percent rpm. From the pump, oil flows through the reverser oil filter, which includes a magnetic chip removing element, to the reverser thrust lever controlled pilot valve, control valve and thrust reverser actuators. Return oil is ported from the control valve case to a "T" fitting in the line between the CSD oil-air cooler and oil-fuel cooler and returned with CSD scavenge oil to the CSD section of the oil reservoir.

Thrust Reverser Indication System

A blue valve IN TRANSIT and amber REVERSE THRUST light are provided for each engine reverse thrust system. These lights, located on the pilots' engine instrument panel, indicate the position of the control valve.

OIL SYSTEM INDICATING ELEMENTS

Quantity, temperature and pressure gages are provided for the engine oil system. Warning lights are provided for both engine and CSD oil systems, and provisions have been made for pre-flight checking of the CSD oil quantity.

Engine Oil Quantity Gages

Four engine oil quantity gages, one for each engine, are located on the flight engineer's instrument panel. Each gage indicates the oil quantity in gallons in the engine section of the oil reservoirs. A dipstick, integral with the gravity fill cap on each engine oil section of the oil reservoir, is available for ground check on the oil quantity.

CSD Oil Quantity Determination

No provisions have been made for in-flight measurement of the CSD section oil reservoir contents. A dipstick, integral with the gravity fill cap on each CSD section of the oil reservoir, is available for ground pre-flight check on CSD section oil quantity.

Engine Oil Temperature Gages

Four engine oil temperature gages, one for each engine, are provided on the flight engineer's instrument panel. Each gage indicates the oil temperature in degrees Centigrade. The temperature sensor probe is located between the engine oil scavenge filter and the engine oil-fuel cooler.

Engine Oil Pressure Gages

Four engine oil pressure gages, one for each engine, are provided on the flight engineer's instrument panel. Each gage indicates the oil pressure in psi units. The oil pressure takeoff point is immediately after the engine oil filter.

Engine Oil Pressure Warning Lights

Four engine oil pressure warning lights, one for each engine, are provided on the pilots' engine instrument panel. The oil pressure takeoff point is immediately after the engine oil filter and the pressure switches are set to predetermined critical pressure settings. The amber OIL PRESSURE LOW lights will illuminate when the critical low pressure setting is reached.

CSD Oil Temperature Warning Lights

Four warning lights, one for each CSD unit, are available for temperature indication purposes in connection with the CSD oil system. The red DRIVE MALFUNCTION lights are located in the electrical section of the flight engineer's instrument panel. The warning light will illuminate when a temperature sensitive probe in the CSD unit detects an oil temperature in excess of 177 degrees C (350 degrees F).

OIL SYSTEM ELECTRICAL SOURCES

For detailed information, consult the WIRING DIAGRAM MANUAL.

The oil pressure gage system operates from 26-volt ac.
The oil temperature gage system operates from 28-volt dc.
The oil pressure warning lights operate from 28-volt dc.
The oil quantity gage system operates from 115-volt ac, 400 cps.
The DRIVE MALFUNCTION warning lights operate from 28-volt dc.

Section 12

HYDRAULIC SYSTEM

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HYDRAULIC SYSTEM

HYDRAULIC POWER SYSTEMS

Hydraulic power is provided by two separate and independent systems, the No. 1 hydraulic power system and the No. 2 hydraulic power system. An electric hydraulic pump acts as an auxiliary hydraulic power system to operate both normal systems. The No. 1 and No. 2 systems are connected only at their reservoirs and through the auxiliary pump pressure supply line. The No. 1 system furnishes hydraulic power for nose landing gear operation, nose wheel steering, nose wheel brakes, the horizontal stabilizer trim system and four fuel jettison pumps. The No. 2 system furnishes hydraulic power for main landing gear operation, main gear wheel brakes and two fuel scavenge pumps. Both systems supply power for the operation of the wing flaps and spoilers.

No. 1 HYDRAULIC POWER SYSTEM

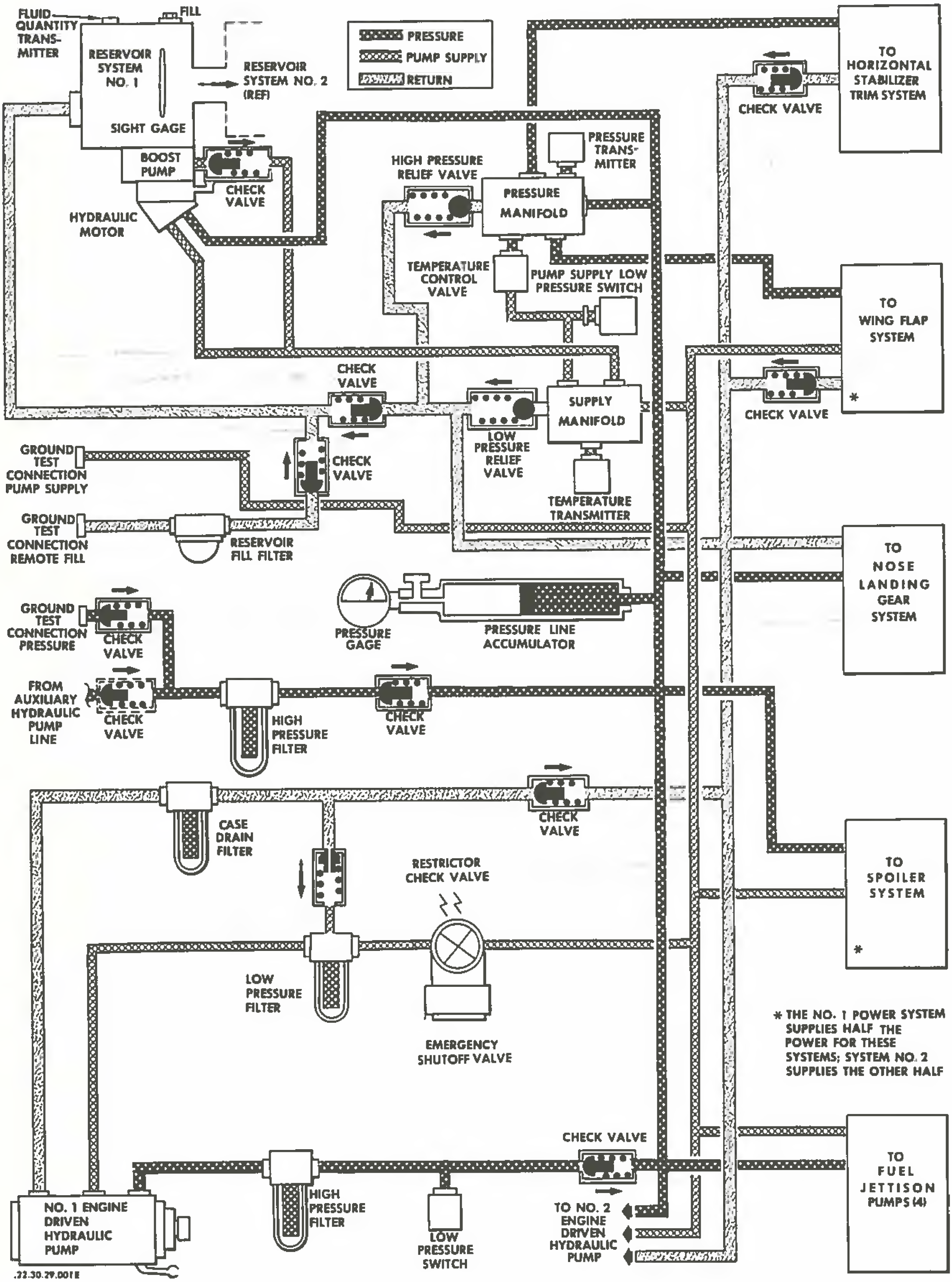
The No. 1 hydraulic power system (see Figure 12-1) consists of a hydraulic reservoir, a pump supply line boost pump, two engine-driven hydraulic pumps on engines 1 and 2, two emergency hydraulic shutoff valves, two low and three high-pressure filters, a pressure accumulator, low-pressure switch, pressure and temperature transmitters, a temperature control valve, pressure relief valves, check valves, and ground test connections. The system uses a pressurized pump supply line and a non-pressurized reservoir return line.

No. 2 HYDRAULIC POWER SYSTEM

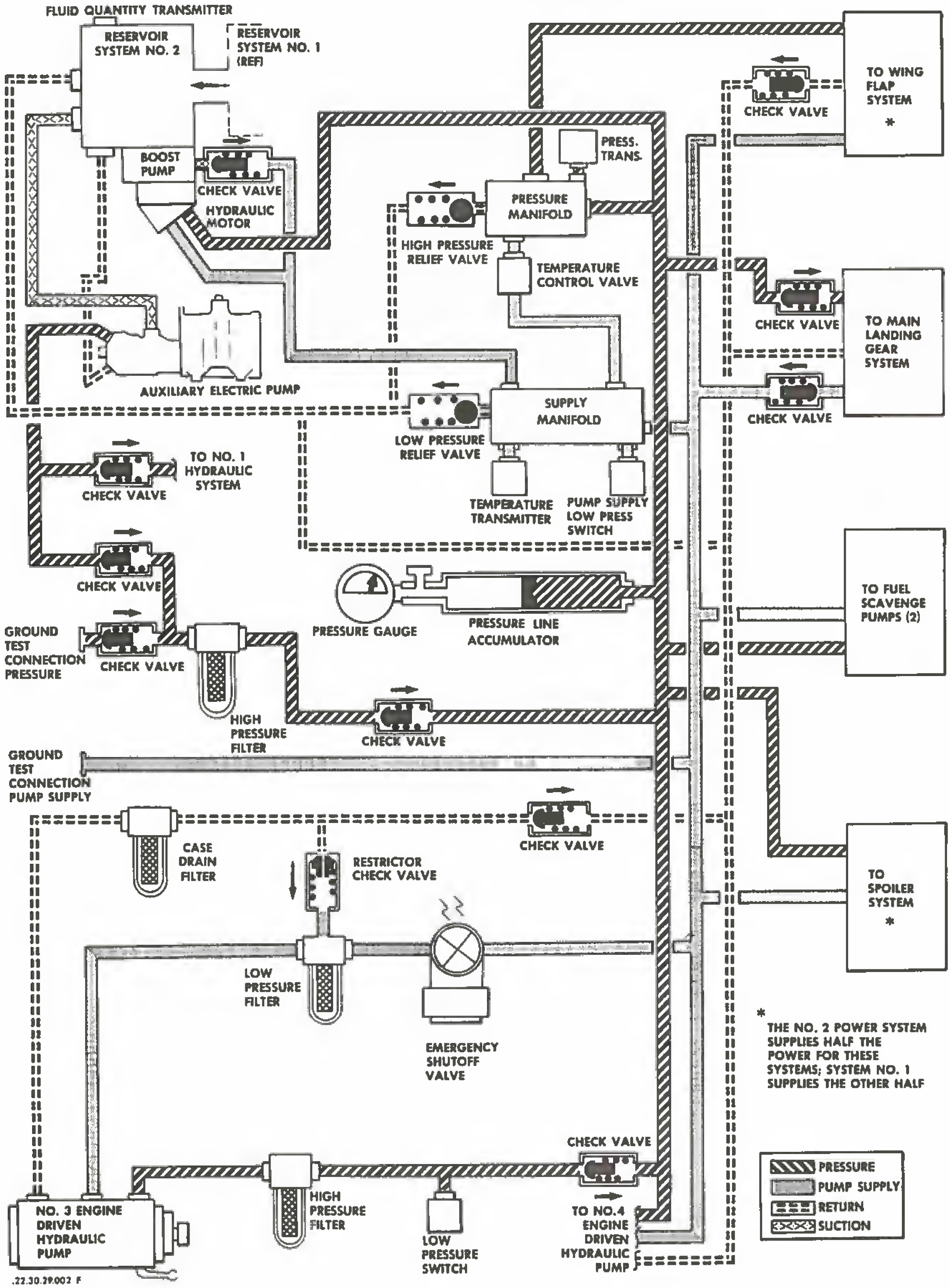
The No. 2 hydraulic power system (see Figure 12-2) is identical in operation and, with few exceptions, employs the same components as the No. 1 system. The hydraulic reservoir in the No. 2 power system is larger and is not equipped with a filler fitting. It is filled simultaneously when the No. 1 reservoir is replenished. The No. 2 reservoir also supplies non-pressurized hydraulic fluid to the auxiliary hydraulic pump. The engine-driven hydraulic pumps for system No. 2 are located on engines 3 and 4.

Reservoirs

Two stainless steel hydraulic reservoirs are used in the hydraulic power systems, one for each system. The No. 2 system reservoir has an indicatible fluid capacity of 6.7 gallons of hydraulic fluid. The No. 1 system reservoir has an indicatible fluid capacity of 2.4 gallons. The reservoirs are located on the right side of the hydraulic and pneumatic compartment. Both reservoirs are non-pressurized and are separately vented to the hydraulic compartment. A fluid connecting line between the two reservoirs permits simultaneous filling of both reservoirs from one filler opening. The connecting line is installed at the reservoir refill level to prevent a leak in either hydraulic system from depleting the fluid in the remaining system below an operational level.



No. 1 Hydraulic Power System
Figure 12-1



No. 2 Hydraulic Power System
Figure 12-2

Reservoir Filler Provisions

There are two methods of filling the reservoirs. A gravity manual fill opening is provided on the top of the No. 1 system reservoir. Both reservoirs can be hand filled through the one opening. Pressure filling is possible through the remote pressure filler connection located next to the hydraulic ground test connections (see Figure 12-3).

Hydraulic Reservoir Sight Gage

A hydraulic reservoir sight gage on the side of the No. 1 system reservoir indicates the fluid level in both reservoirs under three conditions:

1. Zero system pressure with all accumulators discharged.
2. A 3000 psi system pressure is indicated with all accumulators charged and the landing gear down.

Pump Supply Line Boost Pumps

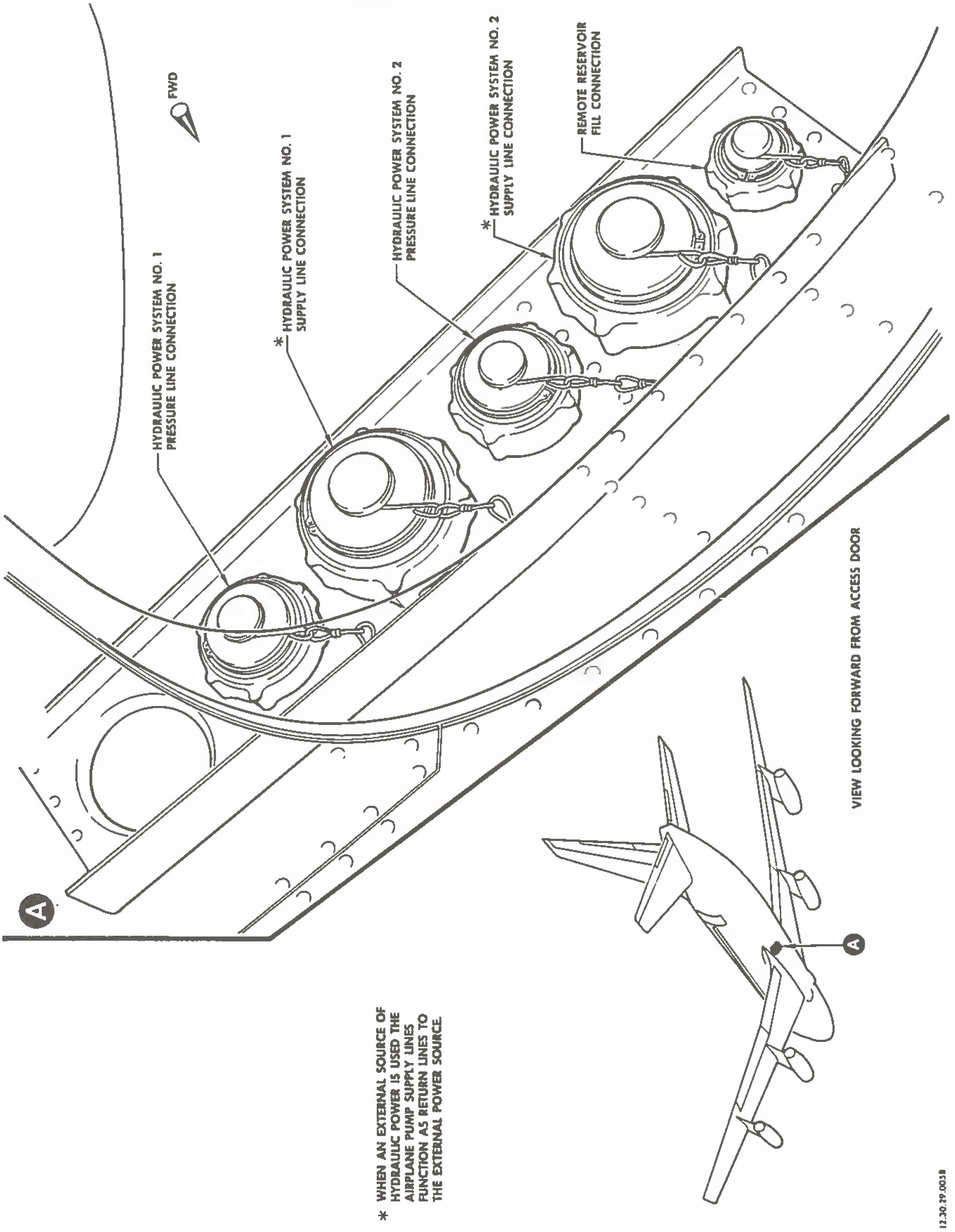
A reservoir pump supply line boost pump is located on the bottom external surface of each reservoir. The boost pumps maintain a nominal 70 to 80 psi pressure in the supply lines to the engine-driven hydraulic pumps. Boost pump motors are operated by the 3000 psi hydraulic system pressure and normally operate in a near stall condition. When a drop in supply lines pressure exists, the pump automatically increases its output to meet the system demand.

Engine-Driven Hydraulic Pumps

Four variable-displacement engine-driven hydraulic pumps are used. One pump is installed on the forward left side of each engine accessory drive. The hydraulic pumps on engines No. 1 and No. 2 supply pressure to the No. 1 hydraulic power system. The hydraulic pumps on engines No. 3 and No. 4 supply pressure to the No. 2 hydraulic power system.

High and Low Pressure Filters

A high pressure filter is installed downstream of each engine-driven pump. These filters have no internal relief provisions. When the pressure across the element exceeds 80 psi differential pressure a red "pop-up" indicator button on the filter raises approximately 1/8 inch. The indicator button remains extended until manually depressed after servicing of the filter element. A low pressure filter is installed between the reser-



Hydraulic System Remote Filler
and Ground Test Connections
Figure 12-3

voirs and each engine-driven pump. These filters are also provided with a red "pop-up" indicator that raises when the pressure differential through the filter exceeds 2.8 to 3.5 psi. A high pressure filter is also installed downstream of the ground test connection for each system.

Pump Inlet Shutoff Valves

A pump inlet shutoff valve is installed in the pressurized supply line to each engine-driven hydraulic pump. These valves are operated by a dc motor. When a FIRE-PULL "T" handle is pulled, the dc motor drives the valve to the closed position and shuts off the hydraulic fluid flow to the engine-driven pump. A guarded and safetied duplicate control switch for each shutoff valve is located on the flight engineer's instrument panel.

CAUTION: PUMP INLET SHUTOFF VALVES SHOULD BE CLOSED ONLY FOR EMERGENCY REASONS. THE ENGINE-DRIVEN HYDRAULIC PUMP WILL BE SERIOUSLY DAMAGED IF OPERATION IS CONTINUED WITHOUT A FLUID SUPPLY.

Pressure Line Accumulators

A pressure line accumulator is installed in each main hydraulic system (No. 1 and No. 2). Each unit is pneumatically charged to 900 psi and dampens surges in the pressure lines. These units also store hydraulic fluid under pressure for momentary use by the operating systems until the pressure from the engine pump is sufficient. The normal operating pressure of the accumulators is 3000 psi. A pressure gage is attached to each of the pressure line accumulators. The range markings are red from 0 to 900 psi and green from 900 psi to 3000 psi. Gage range is from 0 to 5000 psi. The gages indicate the amount of air pressure in the accumulators when the hydraulic pressure is 0 psi. They also indicate the pressure in the hydraulic system when the system is pressurized.

AUXILIARY HYDRAULIC POWER SYSTEM

The auxiliary hydraulic power system supplies pressure to both main systems (No. 1 and No. 2). The auxiliary system is normally used to provide hydraulic pressure for ground service checkout of the controls, and to maintain parking brake pressure.

The auxiliary hydraulic power system consists of an electric hydraulic pump, two filters, and the necessary connections and check valves to connect the system into the No. 1 and No. 2 hydraulic power systems.

Auxiliary Hydraulic Pump (Electric Driven)

The auxiliary hydraulic pump is a three-phase, 115/220-volt, ac operated, variable displacement pump controlled by the HYD PUMP switch on the flight engineer's panel. The pump is located in the hydraulic and pneumatic compartment. The pump operates at 4500 rpm and delivers a minimum of three gallons of hydraulic fluid per minute at 2600 psi. Hydraulic fluid is obtained from the No. 2 system reservoir through an auxiliary suction port located near the reservoir interconnect line level. An inverted standpipe prevents emptying the No. 2 system reservoir in the event the auxiliary pump is operating and a leak develops in system No. 1.

Auxiliary Hydraulic Pump Filters

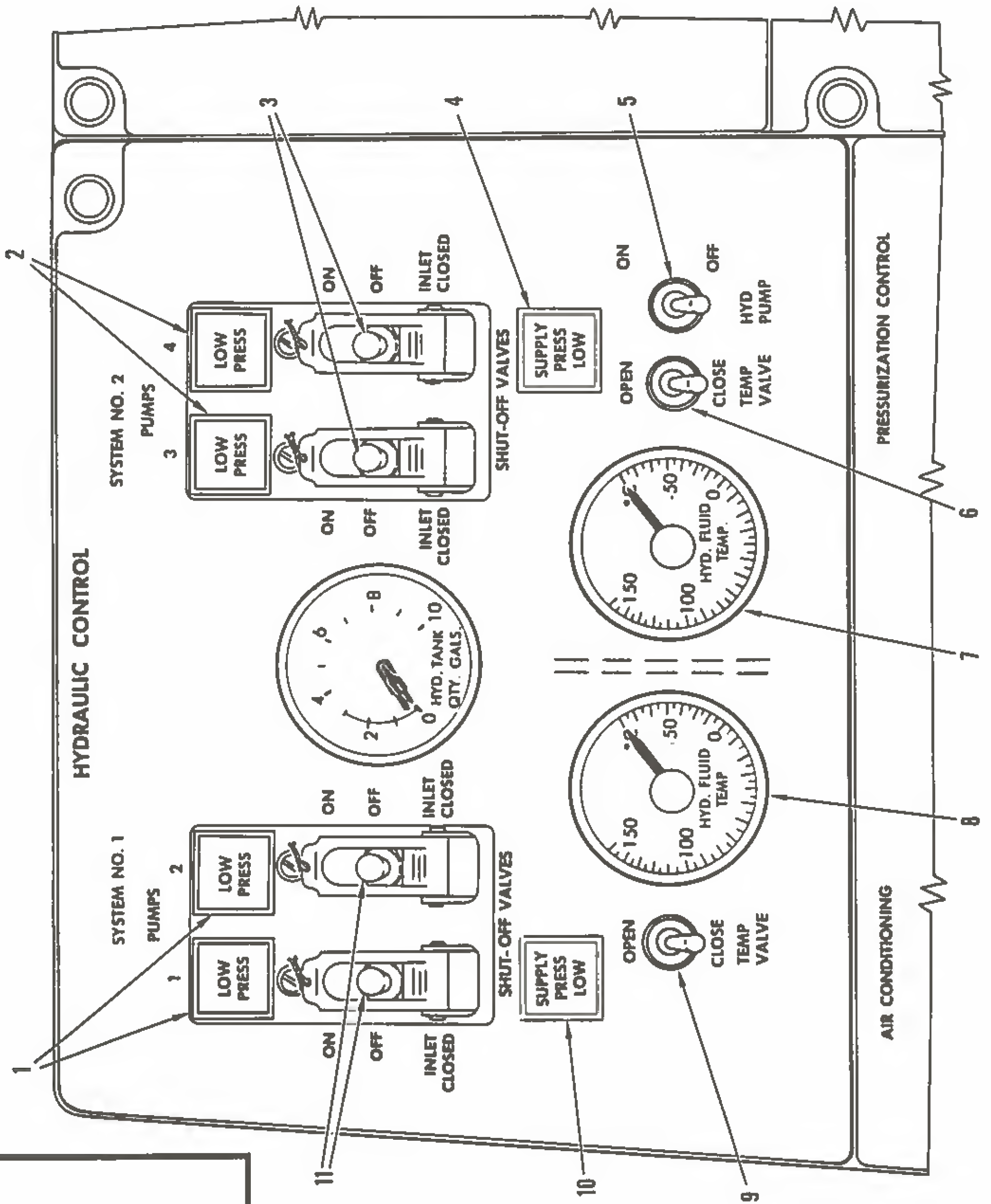
A high-pressure filter is located in the auxiliary pump pressure line to each main hydraulic system . A "pop-up" type indicator on each filter raises 3/16 inch when a differential pressure of 80 psi exists in the filter element. The word "dirty" is exposed with the indicator in the raised position. The indicator must be manually depressed when the filter is serviced.

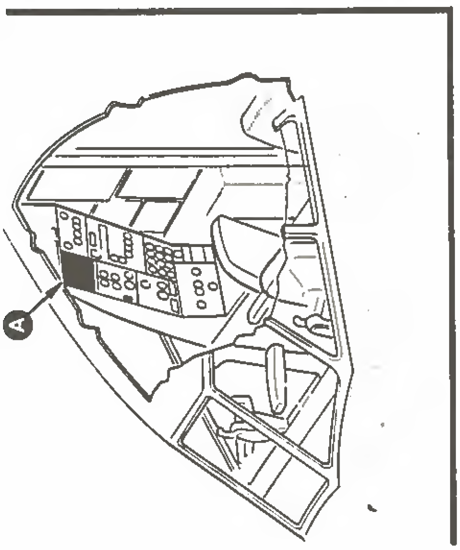
HYDRAULIC SYSTEM CONTROL AND INDICATING UNITS

All No. 1 and No. 2 hydraulic power system control switches and indicators except the pressure indicators for the two power systems and the brake pressure indicator, are located on the upper left portion of the flight engineer's control panel (see Figure 12-4).

System Low-Pressure Switches

A pressure sensitive switch, located in each engine-driven hydraulic pump pressure line, detects excessive low output pressure. Each switch connects to a LOW PRESS warning light located on the flight engineer's panel. The switch closes at approximately 1000 psi.





1. SYSTEM NO. 1 PUMP PRESSURE LINE LOW PRESSURE WARNING LIGHTS.
2. SYSTEM NO. 2 PUMP PRESSURE LINE LOW PRESSURE WARNING LIGHTS.
3. SYSTEM NO. 2 EMERGENCY SHUTOFF VALVE SWITCHES
4. SYSTEM NO. 2 PUMP SUPPLY SYSTEM LOW-PRESSURE WARNING LIGHT.
5. AUXILIARY HYDRAULIC PUMP CONTROL SWITCH.
6. SYSTEM NO. 2 HYDRAULIC FLUID TEMPERATURE CONTROL SWITCH.
7. SYSTEM NO. 2 HYDRAULIC FLUID TEMPERATURE INDICATOR.
8. SYSTEM NO. 1 HYDRAULIC FLUID TEMPERATURE INDICATOR.
9. SYSTEM NO. 1 HYDRAULIC FLUID TEMPERATURE CONTROL SWITCH.
10. SYSTEM NO. 1 PUMP SUPPLY SYSTEM LOW-PRESSURE WARNING LIGHT.
11. SYSTEM NO. 1 EMERGENCY SHUTOFF VALVE SWITCHES

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Flight Engineer's Hydraulic Control Panel
Figure 12-4

Pump Supply Low-Pressure Switches

A pressure sensitive switch is installed in the pressurized supply line to each main hydraulic power system. These switches energize a SUPPLY PRESS LOW warning light on the flight engineer's panel when the line pressure drops to 10 psi.

Remote Hydraulic Pressure Indicators

A hydraulic pressure transmitter in each main hydraulic power system pressure line transmits system pressure indications to their respective dial indicators located on the copilot's instrument panel. The indicators are placarded SYSTEM 1 and SYSTEM 2 and are calibrated from 0 to 4000 psi.

Hydraulic Fluid Temperature Control Valves

A dc-powered hydraulic fluid temperature control valve is installed in each main hydraulic power system. The valves are two-way, two-position, solenoid-operated valves controlled by separate switches on the flight engineer's panel. The TEMP VALVE switches have two positions, OPEN and CLOSE, and normally remain in the closed position. When the valves are open, hydraulic system pressure is routed through a restricted orifice which allows approximately four gallons of fluid per minute to flow from the system pressure line into the pressurized pump supply line. The resulting drop in the main system pressure cause the engine-driven pumps to increase output and thus furnish a supply of warmer fluid throughout the system.

Temperature Transmitters and Indicators

An electric temperature transmitter is installed in the pressurized pump supply line of each main hydraulic power system. The temperature transmitters indicate the hydraulic supply line fluid temperature by means of dial indicators on the flight engineer's panel. The temperature indicators are calibrated from minus 70 degrees to plus 150 degrees Centigrade.

Hydraulic Fluid Quantity Indicators

A fluid level transmitter in each hydraulic reservoir transmits fluid level signals to a dual-pointer indicator on the flight engineer's panel. The pointers are numbered 1 and 2 and the indicator range is 0 to 10 gallons.

HYDRAULIC POWER SYSTEMS ELECTRICAL SOURCES

For detailed information, consult the WIRING DIAGRAM MANUAL.

Emergency Hydraulic shutoff valves.	28-volt dc
Auxiliary hydraulic pump.	115-volt ac, 400 cps
Low-pressure switches	28-volt dc
Pump supply line low-pressure switches.	28-volt dc
Hydraulic system pressure indicators.	26-volt ac
Temperature control valves.	28-volt dc
Temperature indicators.	26-volt ac
Fluid quantity indicators	115-volt ac, 400 cps

Section 13

ELECTRICAL SYSTEM

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ELECTRICAL SYSTEM

GENERAL AIRPLANE ELECTRICAL SYSTEM

The electrical system represents a marked departure from the low-voltage direct-current systems in general use for many years. All electric energy for flight operations is supplied by 115-volt, 400-cycle, 3-phase, ac generators, one mounted on each of the four turbojet engines (see Figure 13-1). Direct current for various airplane requirements is derived from transformer-rectifiers which convert the 115-volt ac generator current to 28-volt direct current. Low alternating-current voltage for instrument use is derived from voltage step-down transformers. A 27.5-volt storage battery provides an emergency source of dc power. External ac power is plugged into the system for use on the ground when the engines are not running.

To maintain constant frequency and also to combine the output of two or more ac generators, it is necessary to operate them at the same rotational speed. This is accomplished by connecting each generator to the engine through a hydraulically operated constant speed drive (CSD).

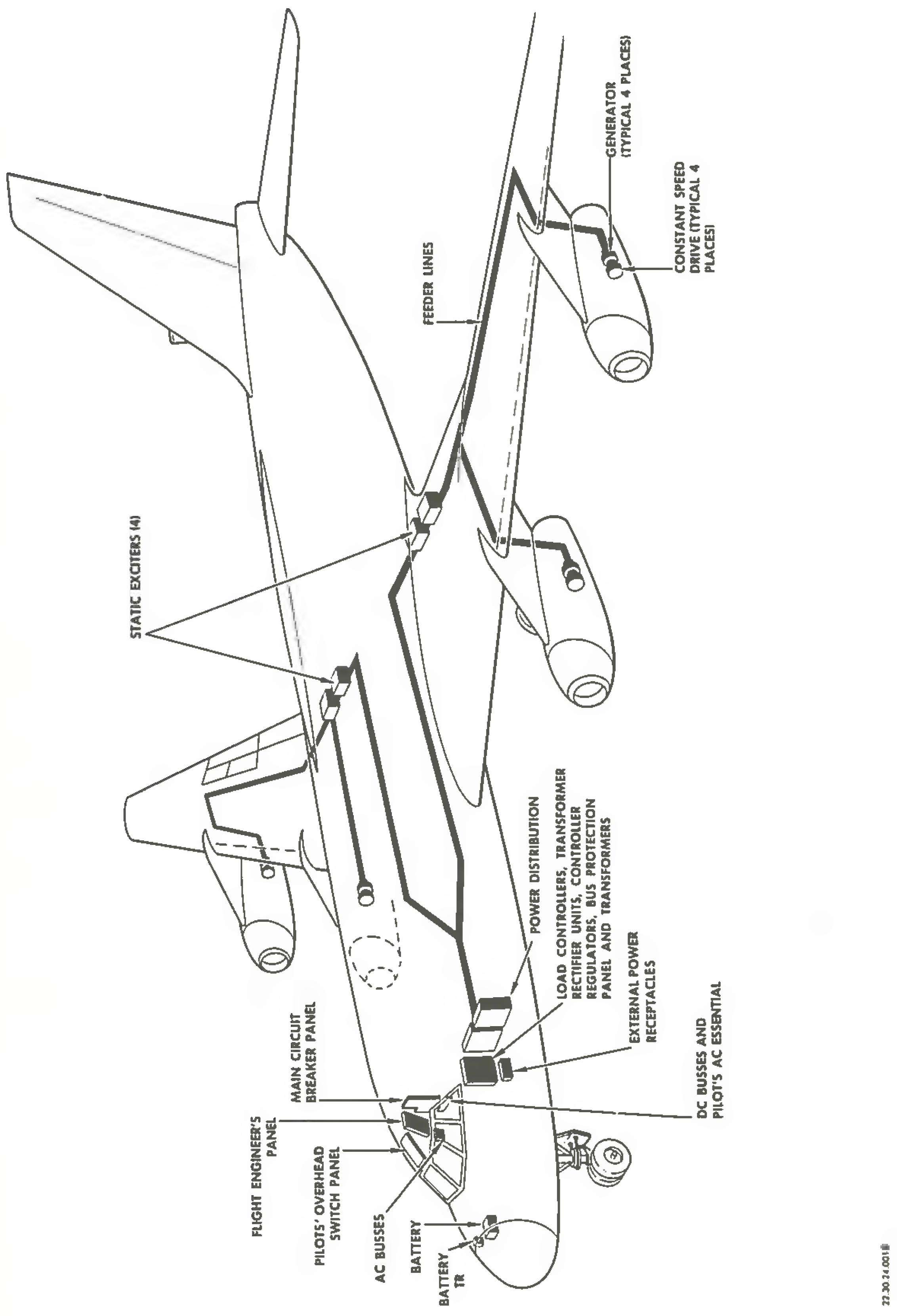
CONSTANT SPEED DRIVE

The output or generator drive speed of the CSD units is monitored and adjusted by the electrical system to keep the generators synchronized, divide the electric load equally among them, and disconnect a generator from the line and shut it down in case of faulty operation.

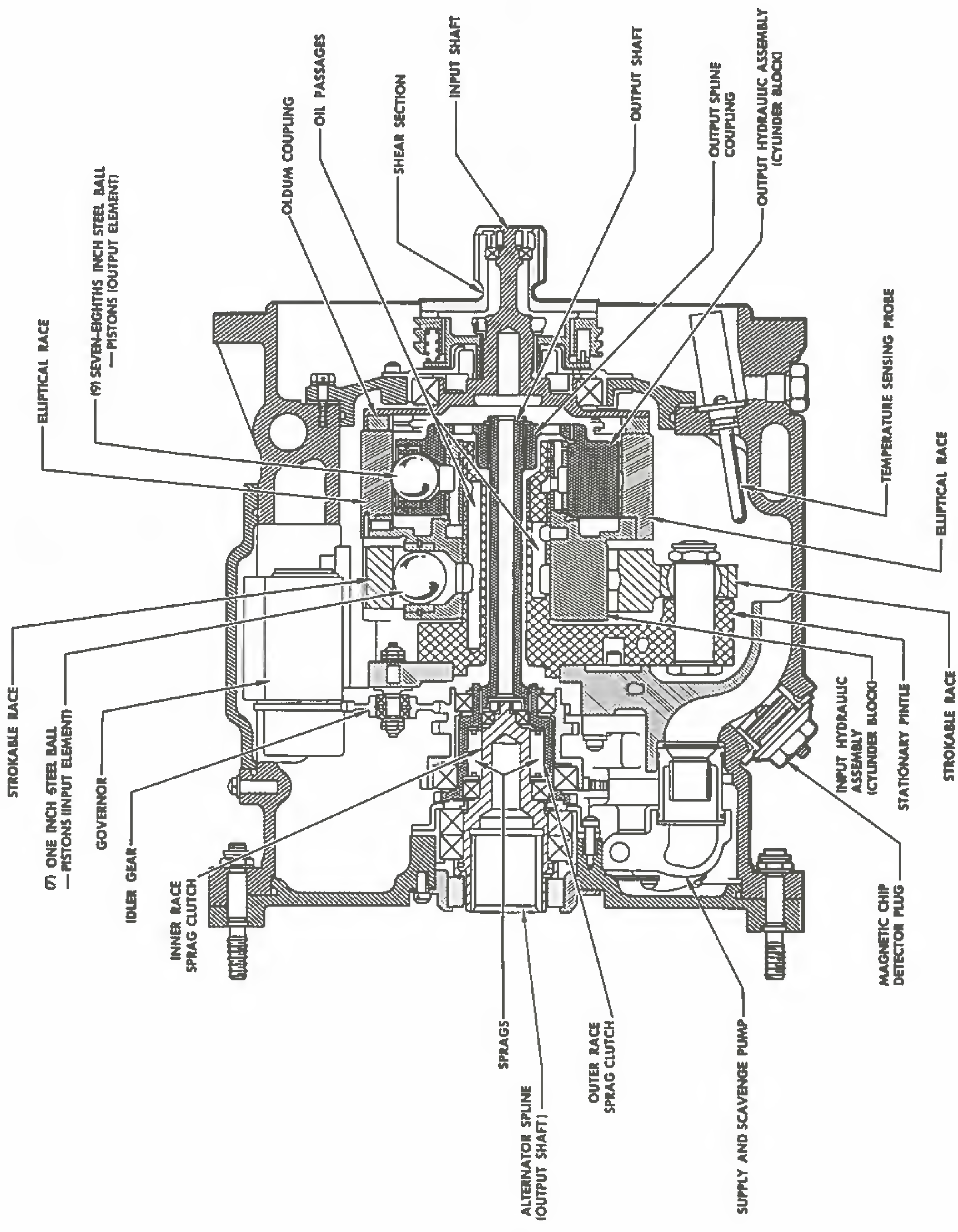
The CSD unit functions as an automatic differential transmission furnishing the driving speed increase or decrease required to maintain a constant generator speed throughout the engine operating range. It will convert any engine drive speed between 4300 and 7760 rpm to a constant 6000 rpm output speed to the generator. This speed corresponds to a line frequency of 400 cycles per second.

The operating mechanism of the CSD consists of a variable volume hydraulic pump connected to the engine drive, a hydraulic motor connected to the generator, two governors, a supply and scavenge oil pump, and electric units and circuits for equipment protection and synchronous control (see Figure 13-2). Each hydraulic unit contains a cylinder block, a tracking race, and ball pistons which are kept in firm contact with the tracking race by centrifugal force. The tracking race of the pump is circular but that of the motor is elliptical. Passages permit oil to flow back and forth between the pump and the motor.

The cylinder block of the pump and the tracking race of the motor form a single unit connected to the engine drive. The tracking race of the pump, which is pivoted to provide for varying the pump displacement, is positioned by the action of the control governor. The cylinder block of the motor is connected to the generator drive. There is no mechanical connection between the engine drive and the generator.



Airplane Electrical Power System
Figure 13-1



CSD Cross Section View
Figure 13-2

When the engine is turning at the speed for which the CSD is set - in this case 6000 rpm - the control governor will maintain the pump tracking race in the neutral position. With the race in this position, there will be no pumping action and the motor block will rotate at the same speed as its tracking race. At lower engine speeds, the governor will move the pump race toward the increase-speed position, high pressure will be supplied to the motor, and the motor block will be driven forward within its rotating race. At higher engine speeds, the governor action will swing the pump race toward the decrease-speed side of neutral, the oil flow through the pump will be reversed, and the motor cylinder block will turn backwards within its rotating race reducing the speed of the generator.

Overspeed-Underspeed Control

An overspeed governor in each CSD unit protects the generator from excessive speed if the primary speed control should fail. If the output speed of the CSD should exceed 7200 rpm, the overspeed governor disconnects the control governor and actuates a hydraulic valve to move the pump race to the full decrease-speed position. Normal operation of the unit cannot be resumed until the engine has been shut down and the oil pressure within the hydraulic unit has dissipated.

A second set of flyweights within the governor operates an underspeed switch. This switch places the generator on the line when the output speed of the CSD increases to between 5625 and 5775 rpm and disconnects the generator when the speed decreases to between 5400 and 5250 rpm.

Mechanical Disconnect

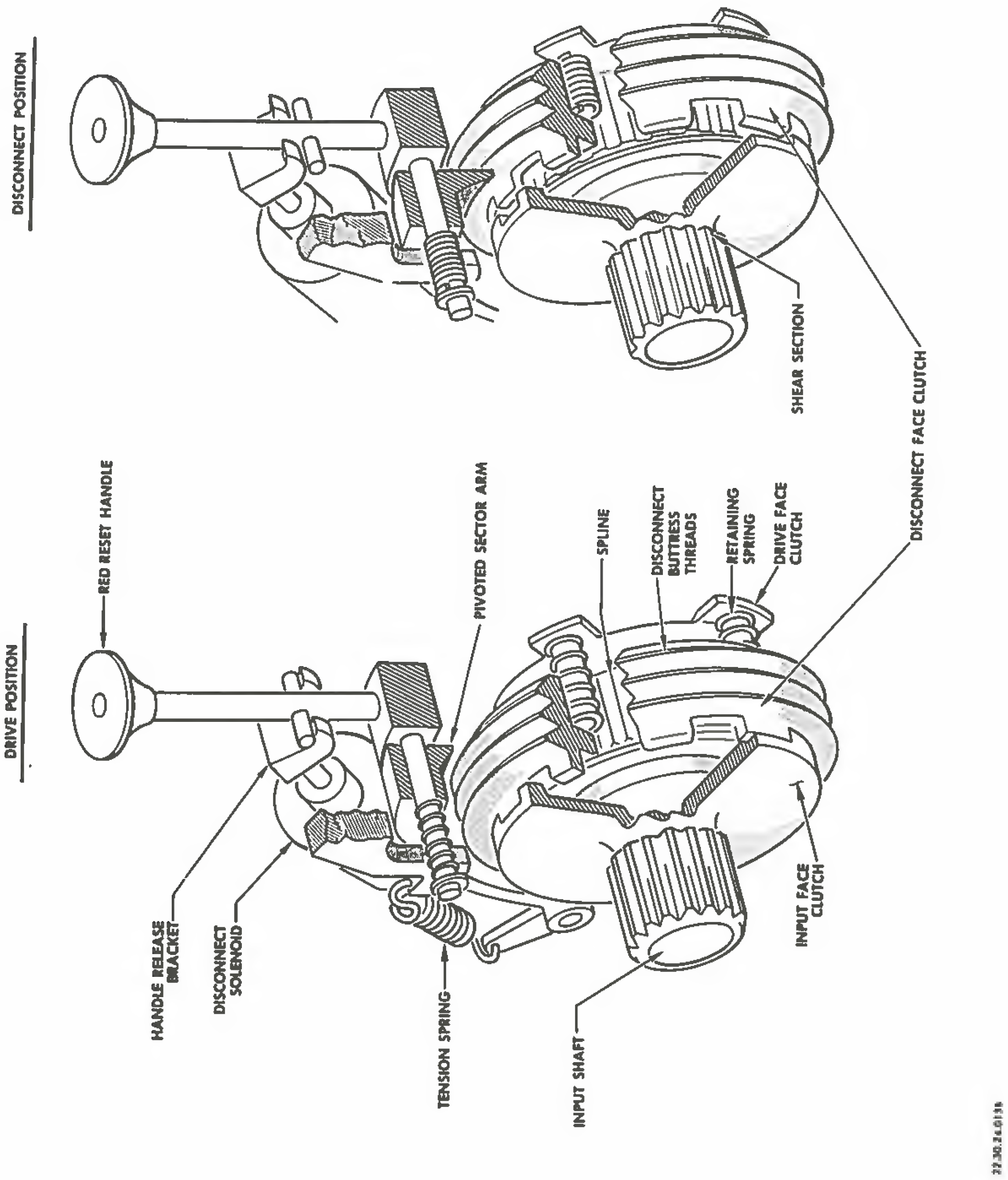
A switch on the flight engineer's panel actuates a mechanical disconnect mechanism, disengaging a face clutch in the drive shaft between the CSD unit and the engine. This clutch cannot be reengaged during flight. Manual reset is required by pulling the reset handle located at the CSD unit (see Figure 13-3).

Overrunning Clutch

An overrunning sprag-type clutch is located between each CSD output shaft and the generator drive. This free-wheeling clutch prevents the generator from driving the CSD if the generator should become motorized during an unbalanced load condition.

Thermal Switch

A thermal switch in the oil sump of each CSD provides an indication of an overheat condition. A red warning light on the flight engineer's panel illuminates when the oil temperature at the switch exceeds 177 degrees C (350 degrees F).



CSD Quick-Disconnect Unit
Figure 13-3

Load-Limiting Switch

A centrifugal-type switch mounted on the same shaft as the engine tachometer, and connected in series with the underspeed switch in the CSD, prevents each generator from imposing a load on its engine during acceleration to IDLE speed. The switch closes when the engine speed exceeds 4250 rpm.

Load Controller

Four load controllers, one for each CSD unit, monitor the in-phase current of each generator and compare it with the average current of all the generators operating in parallel. The load controller modifies the action of the control governor in the CSD to vary the torque output, and correct uneven load distribution (see Figure 13-4).

GENERAL AC ELECTRICAL SYSTEM

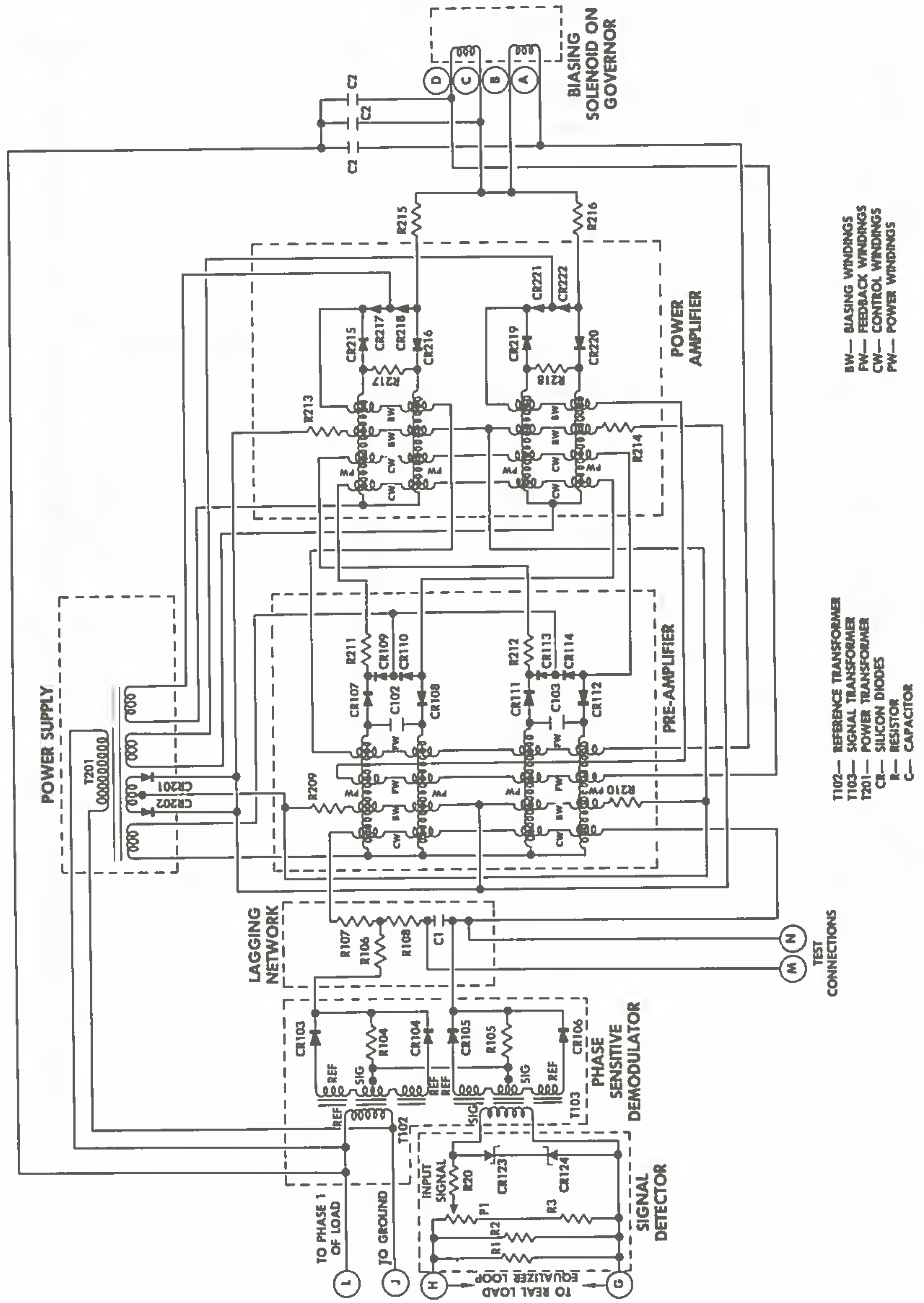
The ac electrical system is a statically-excited four-generator system capable of producing 160 KVA continuously. The output of each generator is fed through its respective static exciter in the trailing edge of the wing, just inboard of the wheel wells. Here, the currents and voltages of the three phases are used to produce excitation for the generator field. From the exciter, the feeders continue through the generator line bus-tie contactor to the load bus.

The ac system contains five load buses: two essential, two nonessential, and a pilot's essential bus. During normal operation, the generators and their respective load buses are paralleled by an automatic paralleling circuit. Bus-tie contactors connect the generator load buses to a synchronizing bus (see Figure 13-5).

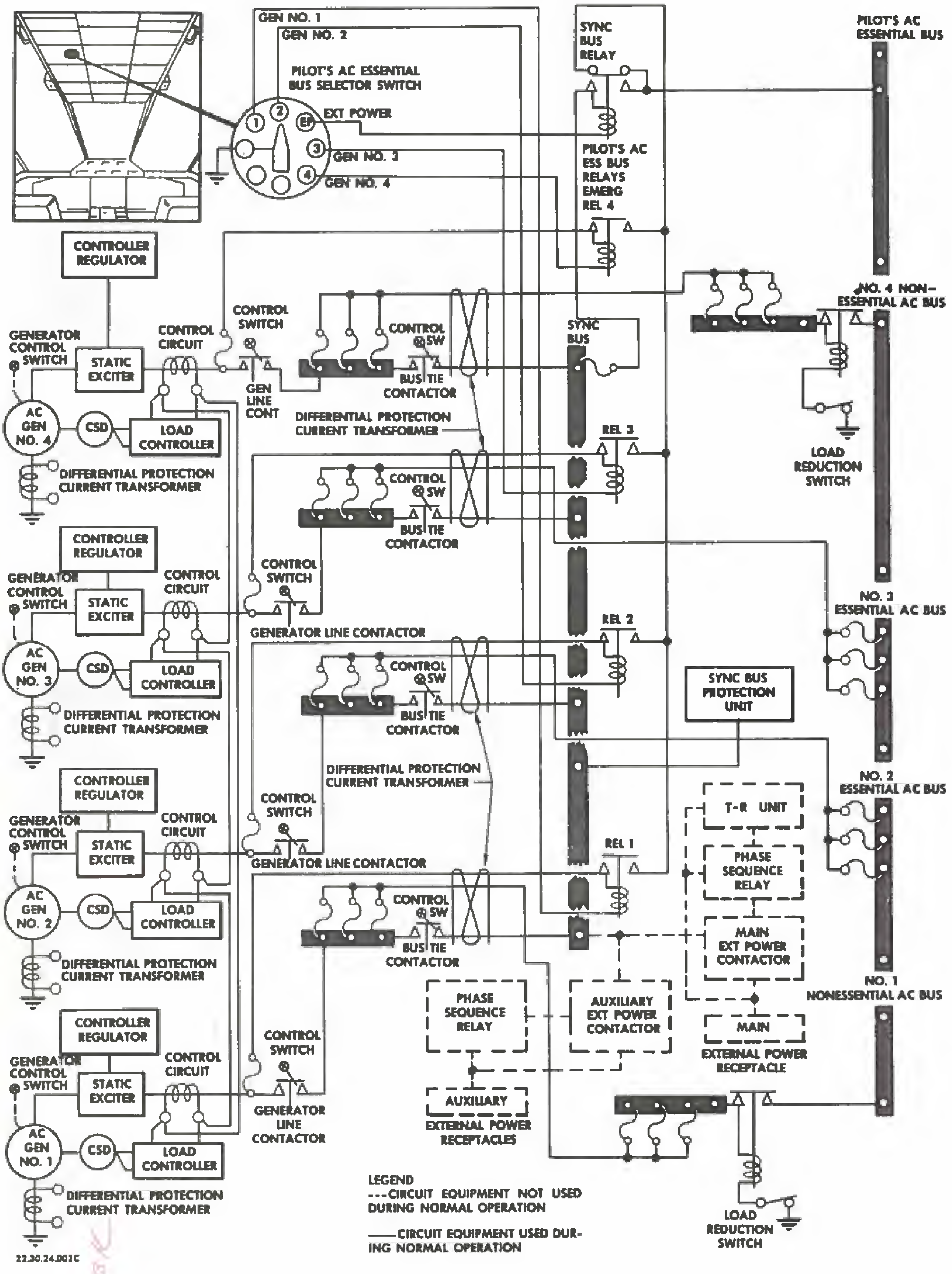
Protective circuits are installed throughout the ac supply system. These protective circuits will automatically isolate a generator and its load bus from the synchronizing bus, or shut down the generator, depending on the fault or malfunction.

Warning lights, instruments, and switches are located on the flight engineer's ac panel for controlling and monitoring the ac electrical system. Through selective switching, a generator can be isolated without loss of electrical power to its load bus. The bus may be supplied electric power by the synchronizing bus.

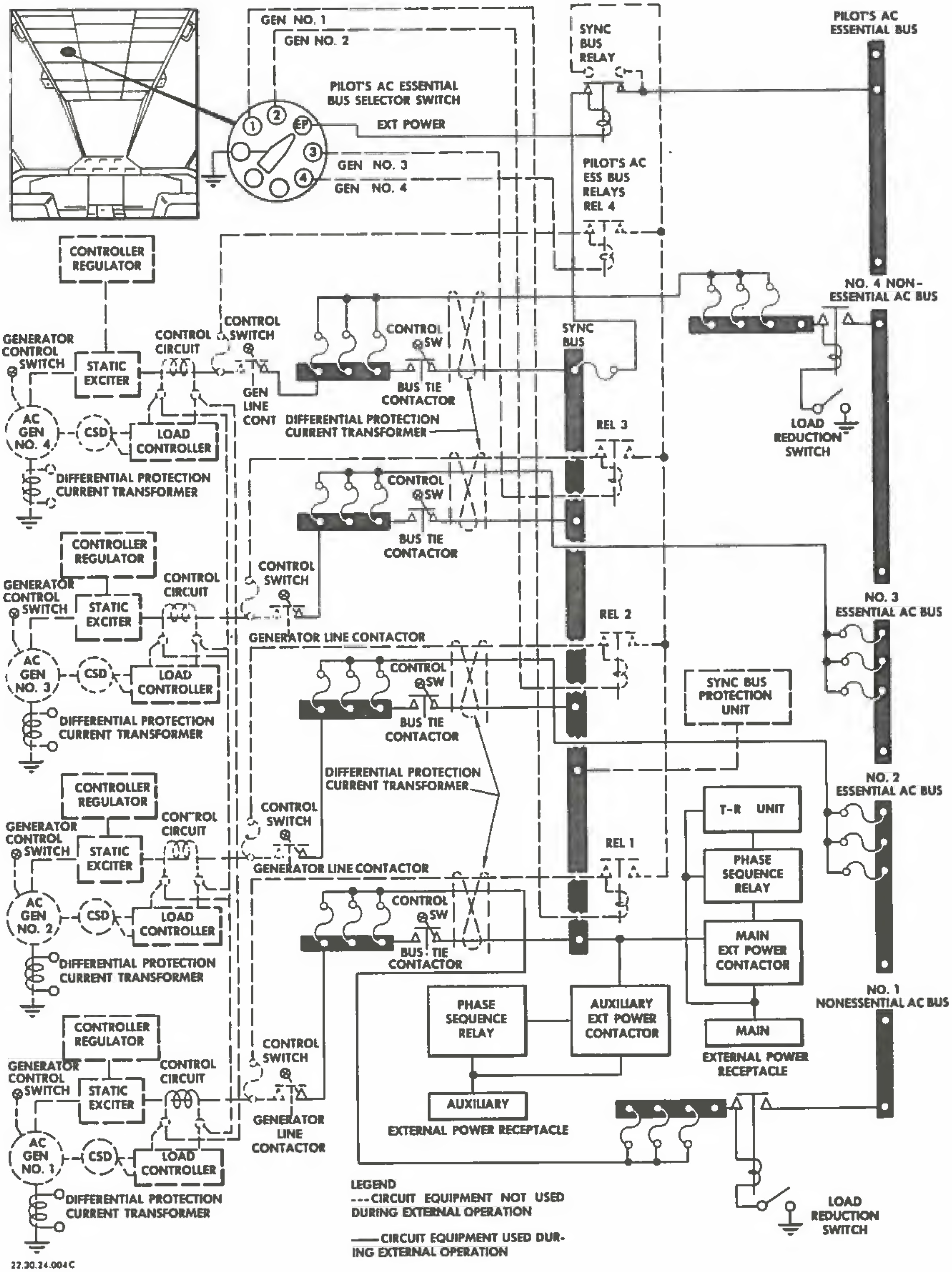
For ground operation, an external ac power supply unit is plugged into the external power receptacle on the right side of the forward fuselage. The airplane's electrical system is protected from damage by a protective circuit which senses the ground power phase and frequency at the plug (see Figure 13-6).



Load Controller Schematic
Figure 13-4



Normal AC Power System
Figure 13-5



External AC Power System
Figure 13-6

AC GENERATORS

Four 120/208-volt, three-phase, Y-connected, 400-cycle, statically-excited ac generators are provided, one in the forward accessory compartment of each engine. Each generator is rated at 40 KVA continuous, with a capability of 60 KVA (150 percent of rated load) for five minutes, and 80 KVA (double load) for five seconds. The generators are self ventilated on the ground by means of integral fans. An air-inlet scoop on the left door of each engine nacelle provides ram air to cool the generators in flight.

Two thermal switches provide generator overheat and generator drive-end warning. One is mounted in the stator end turns and one in the front-bearing support. Overheat actuation of either switch illuminates a warning light on the flight engineer's panel.

Permanent Magnet Generator

A small permanent magnet generator (PMG) rotor,is on the same shaft as the generator rotating field. Its stator provides 52-volt 1600 cps alternating current, which is then rectified to 28-volt direct current for field flashing and control operations. When the generator is producing 115 volts, the PMG is removed automatically as the source of field excitation.

Static Exciter

A static exciter, consisting of saturable-current transformers, rectifiers, and a generator control relay, provides the normal excitation of the generator field. Increase in load results in an increase in exciter output to satisfy the added field current requirements. Conversely, decrease in the load results in a decrease in exciter output.

AC Voltage Regulation

The voltage output of the ac generators is regulated by voltage regulators located in the electrical compartment. The voltage is maintained within 2.5 volts of the rated line-to-ground voltage by varying the dc current from the static exciter to the field winding of the generator.

Reactive Biasing

For optimum performance in parallel generator operation, it is necessary to divide the load current equally between the generators connected to the synchronizing bus. Load current is made up of two components; a real component which is a direct measure of the work done in the system, and a reactive component which is the result of the inductive reactance of the load.

NOTE: This measure of the two components requires a definition of Power Factor. Power factor is defined as the cosine of the phase angle difference between the voltage and the current existing in the system. It may also be considered as the cosine of the angle between the system's apparent power (KVA) and the real power (KW).

The division of real power between the generators in the system is governed by the torque of the constant speed drive. The individual generator control system must sense and control the reactive component of the power. For parallel operation, a reactive load-division circuit is added to bias the voltage regulator to correct for differences in the reactive current between generators.

Overvoltage-Undervoltage Fault Protection System

An undervoltage relay is provided to protect the generator system and load equipment from sustained undervoltages or underexcitition. If an undervoltage condition (96± 4 volts) exists on one or more phases for a period of five to ten seconds, the undervoltage relay will automatically disconnect the generator from its load bus.

The overvoltage relay protects the equipment from sustained overvoltages as well as high circulating currents caused by overexcitation. If the overvoltage relay detects an overvoltage condition in one or more phases, it will automatically disconnect the generator from its load bus after a time delay inversely proportional to the magnitude of overvoltage.

Generator/Feeder Faults

A differential protection relay is installed to remove a faulty generator from its load bus if any type of feeder or generator-stator short circuit occurs. A line-to-line or line-to-neutral current differential of 35 amperes or more between current transformers in the same phase will cause the differential current relay to be energized and the generator to be deenergized and disconnected from its load bus.

Unbalanced Current

An unbalanced current circuit, installed in each generator system, consists of a current transformer loop in phase "c" of each generator, a full wave bridge rectifier, a resistor, and an unbalanced current relay. The unbalanced current relay senses and protects the generator against loss of real load division by the constant speed drive. It is effective during parallel operation and provides back-up protection against failure of the control governor.

Loss of real load division will isolate the faulty generator in a paralleled system. When the current in phase "c" of the generators differs from phase "a" of "b" by more than 50 amperes, the unbalanced current relay will isolate this particular system, and the other systems will not be affected because of the reduced magnitude of the error signal. After isolation, the other protective circuitry will deenergize the faulty generator.

Neutral Current and Open Phase

Neutral-current sensing provides a means of detecting an open phase in the system, in addition to detecting a line-to-ground or a line-to-line-to-ground fault during parallel operation. The circuit for neutral-current sensing makes use of the same current transformer loop used for differential-current protection, except for an additional transformer in the neutral leg of the current transformer's secondary loop. For normally balanced 3-phase loads, the neutral leg does not carry current since the series aiding currents are restricted to the interphase loops.

When an open phase exists, current is produced in the neutral leg because of the missing signal in the tranformer secondaries. The neutral current will flow in the primary of the transformer and energize the neutral-current relay, thus isolating one or more generators from parallel operation. Once the faulty generator has been isolated, it is the function of undervoltage sensing to trip the system.

Load Faults

With 50 to 80 amperes of neutral-line current resulting from a line-to-ground or from a line-to-line-to-ground fault, sufficient current will flow in the primary of the transformer to energize the neutral-current relay. The prime function of the neutral-current relay is to initiate action to split the system so that each generator assumes its own load (assuming that the load limiters have not cleared). Should the fault remain in the system once it is in isolated operation, the undervoltage relay will disconnect the affected generator.

LOAD DISTRIBUTION

There are three types of load buses which provide for ac power distribution: essential, nonessential, and pilot's essential bus. For detailed information, consult the WIRING DIAGRAM MANUAL.

Essential Buses

Essential buses are those required for safe flight. The essential buses are powered by No. 2 and No. 3 generators in isolated operation. If, during parallel operation the No. 2 or No. 3 generator fails, power for the essential bus is provided by the other generators through the synchronizing bus. The failed generator switch is turned off. The load will remain connected to the synchronizing bus as long as the bus-tie contactor switch is closed.

Nonessential Buses

The nonessential buses supply power to systems not required for safety of flight. The No. 1 and No. 4 generators power the nonessential buses in isolated operation. However, the nonessential buses may be energized by the other generators if No. 1 and No. 4 generators fail. The failed generator switch is turned off and the bus-tie contactor switch is left in the closed position.

Synchronizing Bus

The synchronizing bus is utilized for paralleling the generators, and for energizing an individual load bus when its generator has failed. This bus also supplies power to the pilot's essential bus for external power operation.

Pilot's Essential Bus

The pilot's essential bus is powered normally by any one of the four generators selected on the pilot's ESSENTIAL BUS SELECTOR panel. This power is taken from the generator before the line contactor. Thus, it is possible to have a generator operating and supplying power only to the pilot's essential bus. The pilot's essential bus supplies power to the essential flight instruments, to a transformer-rectifier for emergency 28-volt dc power, and to a second transformer-rectifier for battery charging (see Figure 13-7).

26-Volt AC Instrument Power

The 26-volt ac instrument power is delivered from a main or standby 26-volt ac single-phase, stepdown transformer. A two-position 26 V.A.C. POWER switch on the flight engineer's control panel controls the selection. The MAIN 26-volt transformer receives its power from the No. 3 essential bus and the STANDBY transformer from the pilot's essential bus.

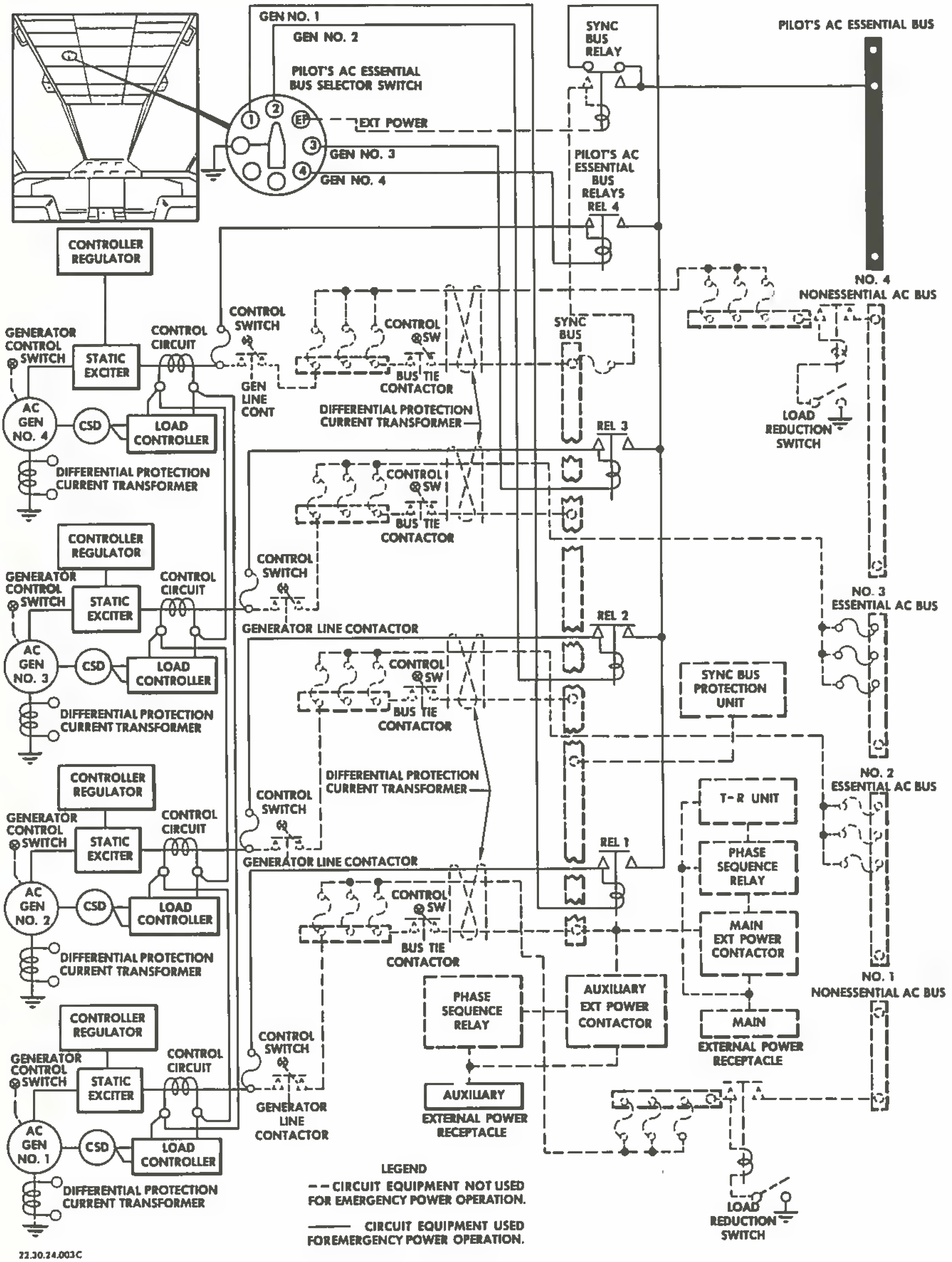
AUTOMATIC-PARALLELING

The automatic-paralleling circuit ties each generator to the synchronizing bus at the proper instant to minimize system disturbances. The automatic-paralleling relay will synchronize the generators only when they are in the on-frequency range. It will not bring the generators within the on-frequency range. The generator output frequency is controlled by the governor in the constant speed drive.

To synchronize generators, momentarily actuate the EXTERNAL POWER switch to the GEN PARALLEL position, thereby connecting one system to the synchronizing bus through the sequence-switching circuits. Each of the remaining systems will automatically be connected to the bus by its auto-paralleling relay.

Auto-Paralleling Relay

The auto-paralleling relay consists of a full-wave bridge circuit, an auto-paralleling relay, and a reference diode. The bridge circuit receives two signals: one from phase "c" of the synchronizing bus through a current limiting resistor, and a second from phase "c" of the system to be paralleled. If the two signals applied across the bridge network are at different frequencies, the voltage difference will vary as the difference in frequency. When the modulation (difference voltage) signal has reaches the 90 degree point, the magnitude of the difference in voltage impressed is sufficient to trigger the breakdown diodes. The auto-paralleling relay is then actuated until the 270 degree point is reached, beyond which the reference diode ceases conducting and the relay will drop out. The drop-out initiates the paralleling action.



Emergency Operation of the AC System
Figure 13-7

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B

If the difference in frequency is less than 8 cps, there will be sufficient elapsed time to close the various contactors necessary to parallel. If the difference is greater than 8 cps, the auto-paralleling relay will energize before the contactors can close and the generator will not parallel.

The maximum phase difference between the synchronizing bus and the generator to be paralleled is 90 degrees. Beyond 90 degrees the system will not parallel.

Paralleling Sequence

The No. 1 generator normally is connected to the bus first and the other generators are paralleled to it. If, for any reason, the No. 1 generator does not go on the bus but remains isolated, the next generator in sequence (the No. 2 generator) will be connected to the bus and the other generators will parallel to it. If the No. 2 generator is not ready to go on the bus, then the No. 3 generator will be connected to the bus and it in turn will be paralleled by the No. 4 generator.

For example, if the No. 1 generator is connected to the bus and the No. 2 generator will not come on the line because it is out of phase with the No. 1 generator by 100 degrees, then the No. 3 generator will connect to the bus and the No. 4 generator will parallel it. When the No. 2 generator gets back in phase with the No. 1 generator, or the synchronizing bus, it will then parallel.

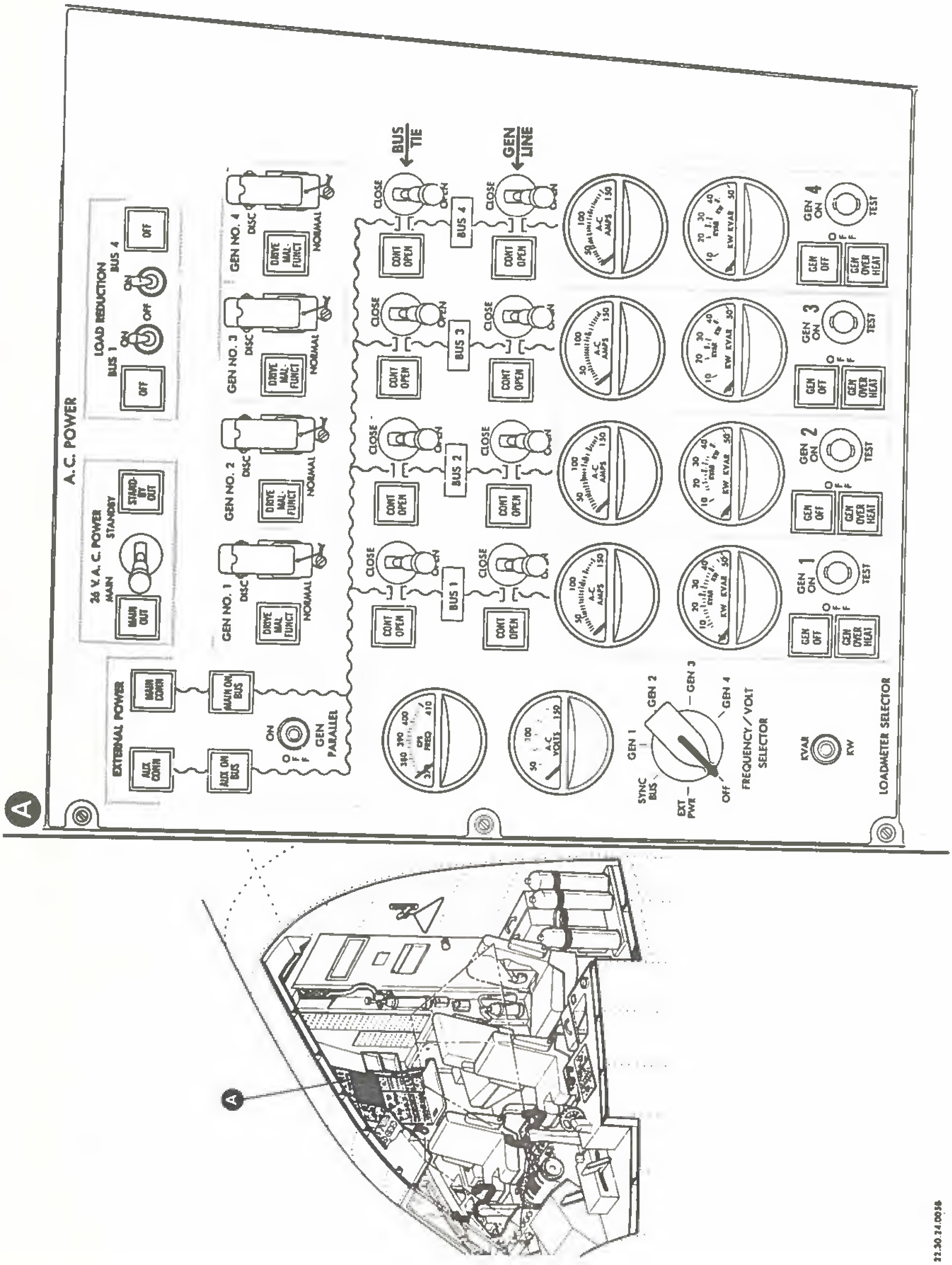
FLIGHT ENGINEER'S AC CONTROL PANEL

The operation of the ac system is controlled through the ac control panel located at the flight engineer's station (see Figure 13-8). With the exception of the pilot's essential bus, this control panel is equipped with the necessary warning lights, instruments, and switches required for complete control of the ac system. The selector for the pilot's essential bus power source is on the pilot's overhead switch panel.

The ac system is operated automatically with all switches in the ON or CLOSE position. The system may be isolated by the action of the protective units or at the discretion of the flight engineer. The switches and warning lights are grouped to illustrate the flow and contol of the power. There is a generator-control switch, a generator line-contactor switch, a bus-tie contactor switch, and a generator disconnect switch for each generator.

Generator Control Switch

Each GEN control switch is a three-position, ON-OFF-TEST, safety-type switch. The handle must be pulled out before the switch can be moved to the ON or OFF position. The TEST position is a spring-loaded momentary contact. In the ON position, the generator is allowed to build up and produce power. In the OFF position, the field of the generator does not receive excitation and the generator output is less than one volt (as a result of residual magnetism). The output of the generator can be checked by setting the selector switch to connect the generator to the voltage and frequency meters and holding the generator control switch in the TEST position.



Flight Engineer's AC Control Panel
Figure 13-8

Generator Line Switch

Each GEN LINE contactor switch is a two-position, OPEN-CLOSE, safety-type switch. The handle must be pulled out to move the switch to the OPEN or CLOSE position. The CLOSE position connects the generator to its load bus; the OPEN position disconnects the generator from its bus.

Bus-Tie Switch

Each BUS-TIE contactor switch is a two-position, OPEN-CLOSE, safety-type switch which must be pulled out to move it to either position. This switch controls the circuitry for opening or closing the contactor between the individual load buses and the synchronizing bus.

Generator Disconnect Switch

Each GEN No. (x) disconnect switch is a guarded two position, NORMAL-DISC, switch. Actuation of the switch to DISC supplies power for disconnecting the constant speed drive input shaft from the engine.

External Power Switch

The EXTERNAL POWER switch is a three-position, ON-OFF-GEN PARALLEL, toggle switch. In the ON position, without the engines operating, external power is supplied to the synchronizing bus and, through the bus-tie connectors, to the load buses. With the engines operating and the switch in the ON position, external power is supplied to the synchronizing bus only. In the OFF position, no external power is applied to the buses. In the GEN PARALLEL position, external power is disconnected and the generators share the loads through the synchronizing bus.

Load Reduction Switch

Two LOAD REDUCTION, two-position, ON-OFF, toggle switches are provided. These switches control the No. 1 and No. 4 nonessential buses. In the ON position, ac power is supplied to the No. 1 or No. 4 nonessential bus, depending on the position of the corresponding switch. In the OFF position, those units operating from the Nonessential bus will be inoperative. In the OFF position, the respective generators still supply power to the other buses through the synchronizing bus.

26-Volt AC Power Switch

The 26 V.A.C. POWER switch is a two-position, MAIN-STANDBY, toggle switch. Actuation of the switch determines which transformer will supply the 26-volt ac instrument bus.

KW-KVAR Selector Switch

The KW-KVAR selector switch is a two-position KW-KVAR toggle switch. The four KW-KVAR meters read kilowatt load or kilovolt-ampere reactive load depending on the position selected.

Frequency-Volt Selector Switch

The FREQUENCY/VOLT SELECTOR switch is a seven-position switch allowing the flight engineer to read the frequency and voltage of the external power unit, the synchronizing bus, and each individual generator. An OFF position is provided.

AC SYSTEM WARNING LIGHTS

Warning lights are located on the flight engineer's ac instrument panel to warn of malfunction and indicate the position of the relays that control the operation of the ac system.

Generator Overheat Lights

Four red GEN OVERHEAT lights, one for each generator, are located along the bottom of the ac panel. A light will illuminate if a generator becomes overheated due to drive-end bearing friction or electrical load.

Generator Off Lights

Four amber GEN OFF lights, one for each generator, illuminate when the generators are not producing power and the dc emergency bus is energized.

Generator Line Lights

Four amber generator line CONT OPEN lights, one for each generator, illuminate when the generator line contactors are open and generator power is not connected to the load buses.

Bus-Tie Lights

Four amber bus-tie CONT OPEN lights, one for each generator, indicate the position of the bus-tie contactor for each generator. A light illuminates when the associated bus-tie contactor is open.

Drive Malfunction Lights

A red DRIVE MALFUNCT light is provided for each generator. A light will illuminate when the constant speed drive of the associated generator becomes overheated.

External-Power Lights

Two white lights, MAIN CONN and AUX CONN, illuminate when the main and the auxiliary power lines are connected to the airplane. Two blue lights, MAIN ON BUS and AUX ON BUS, illuminate when main and auxiliary power is connected to the synchronizing bus.

Load-Reduction Lights

An amber load-reduction OFF light illuminates when No. 1 or No. 2 nonessential buses are disconnected.

26-Volt AC Power Lights

Two 26-volt ac power MAIN OUT and STANDBY OUT warning lights indicate the loss of 26-volt ac power. The main transformer indicator illuminates amber and the standby transformer indicator illuminates red upon failure of the respective unit.

AC METER SYSTEM

Ammeters, KW-KVAR meters, a frequency meter, and voltmeter, all located on the ac instrument panel, enable the flight engineer to monitor the operation of the ac electrical system.

KW-KVAR Meters

Four KW-KVAR meters, one for each generator, provide individual readings for each generator system. The KW-KVAR selector switch controls all four meters. With the selector switch in the KW position, all meters read the real power of their systems in kilowatts. The KVAR position is used to read the reactive power of each system.

Ammeters

Four ac ammeters indicate the current output of each generator.

Frequency Meter

A single frequency meter is used to check the frequency of the external power, the synchronizing bus, and each individual generator. The frequency-volt selector switch controls the desired selection.

Voltmeter

A single ac voltmeter is provided to check the voltage of the external power, the synchronizing bus, or each individual generator. The voltmeter selection is accomplished simultaneously with frequency meter selection.

CIRCUIT PROTECTION

All circuits are protected by current limiters, fuses, or circuit breakers. The circuit breakers are located at the flight engineer's station. The fuses and current limiters are located in the electronic and electrical areas, and at the flight engineer's station. For detailed information, consult the WIRING DIAGRAM MANUAL and Chapter 24 of the MAINTENANCE MANUAL.

DC ELECTRICAL SYSTEM

The dc system does not use separate dc generators but obtains direct current through the use of transformer-rectifiers operating on power from the ac system. The dc system is automatic in operation when all transformer-rectifier (T-R) control switches and the battery switch are in their normal positions.

The dc system is composed of transformer-rectifiers, reverse current relays, a battery relay, an essential bus, an emergency bus, and a 27.5-volt 13.5 ampere-hour battery (see Figure 13-9).

Transformer-Rectifiers

Four 50-ampere transformer-rectifiers are installed in the right electrical equipment rack. The No. 1 main T-R unit is powered from the No. 1 nonessential ac bus, the No. 2 unit from the No. 2 essential ac bus, the No. 3 unit from the No. 3 essential ac bus, and the No. 4 unit from the pilot's essential ac bus. The output voltage range of the T-R units is 24.0 to 27.5 volts for an individual load current range of 0 to 50 amperes. Outputs are paralleled during normal operation, thereby providing a 200-ampere, 28-volt dc bus. The T-R units are of the fan-cooled silicon rectifier type.

Battery

A 27.5 volt, 13.5 ampere-hour (at a 2-hour rate) nickel-cadmium battery is installed in the forward nose section of the airplane immediately inside the nose access door.

Battery Charger

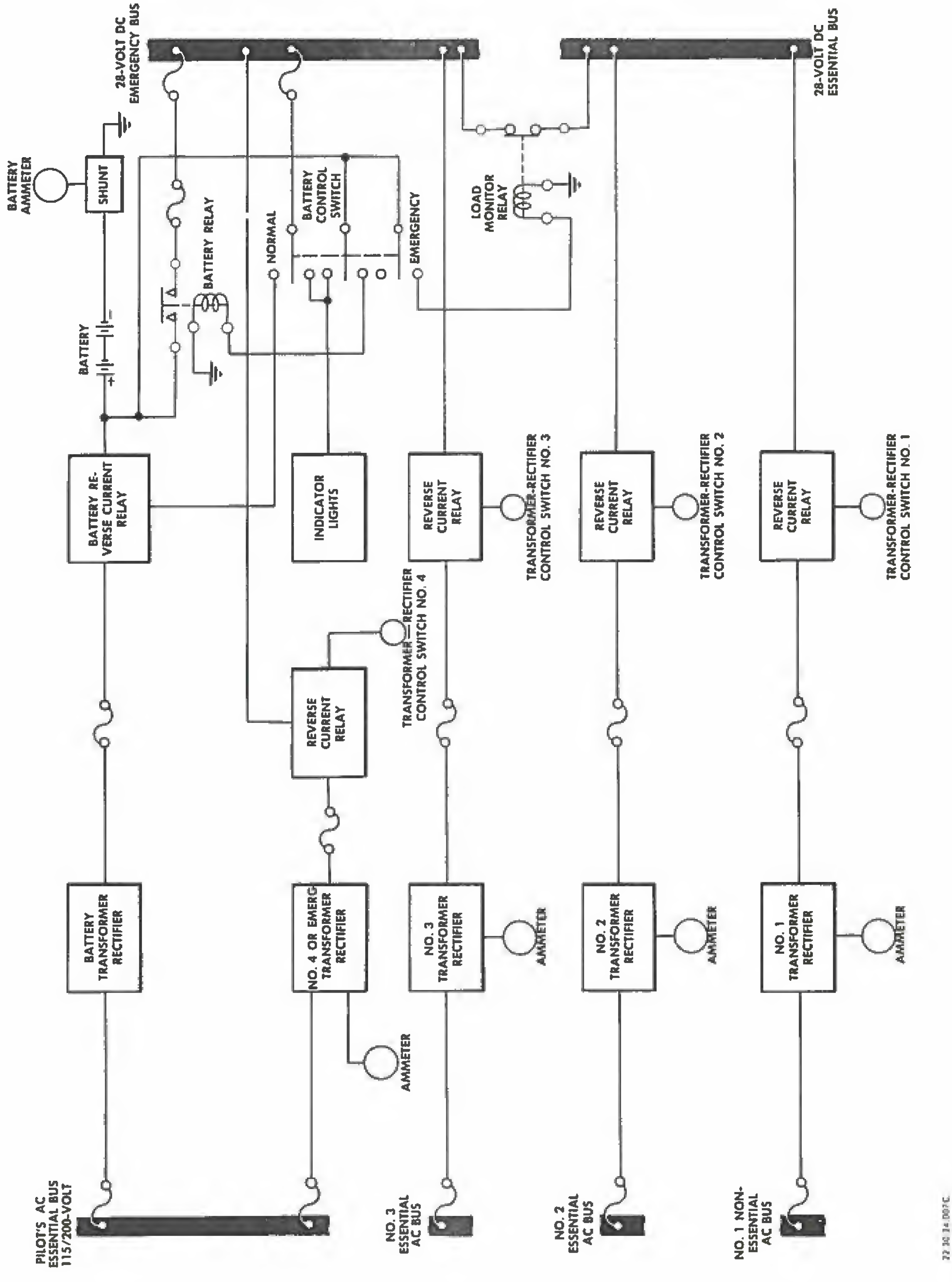
The battery charger is a 29.25 to 33.25-volt, 0 to 20-ampere T-R unit of convection-cooled silicon rectifier design and is powered from the pilot's ac essential bus. It has an overload capacity of 30 amperes for 5 minutes. The unit is installed adjacent to the battery in the forward nose section of the airplane.

Reverse-Current Relays

Identical reverse-current relays are installed in the output lines of each main T-R unit and the battery charger. Those in the main T-R unit outputs serve as bus and wire protection against faults, and also as output control relays. The relay in the battery-charger output provides battery and wire protection against faults, and serves as the battery charger output control relay. The main T-R unit reverse-current relays are located at the dc buses in the dc junction box at the lower aft side of the flight engineer's station. The battery charger reverse-current relay is installed in the battery relay box adjacent to the battery.

Battery-Emergency Relay

A 50-ampere relay, adjacent to the battery, connects the battery to the emergency bus for emergency operation.



DC Power System
Figure 13-9

Essential Bus Disconnect Relay

A 200-ampere capacity relay isolates the essential bus from the emergency bus to prevent excessive current drain on the battery during emergency operation, It is located in the dc junction box at the lower aft side of the flight engineer's station.

Current Limiters

A 50-ampere current limiter is installed adjacent to, and in the output of, each main T-R unit for protection of the wires to the main T-R unit reverse current relays. A 20-ampere limiter is installed adjacent to, and in the output of, the battery charger to protect the wire to the battery charger reverse-current relay. Three groups of three 5-ampere limiters, located in the ac power distribution box at the base of the flight engineer's station, protect the wires to the inputs of No. 1, No. 2, and No. 3 main T-R units. The input wires to the emergency main T-R unit (No. 4) and the battery charger are protected by 5 ampere limiters installed on the pilot's essential ac bus adjacent to the dc junction box. The triple wire feeder through which the battery supplies the emergency dc bus is protected by 30-ampere limiters. Various small limiters protect wires supplying power from the battery bus.

Battery Bus

The battery bus, located in the battery relay box, supplies power to the battery control switch, the inertia emergency lighting circuit, and the landing gear warning horn circuit.

DC Power Distribution

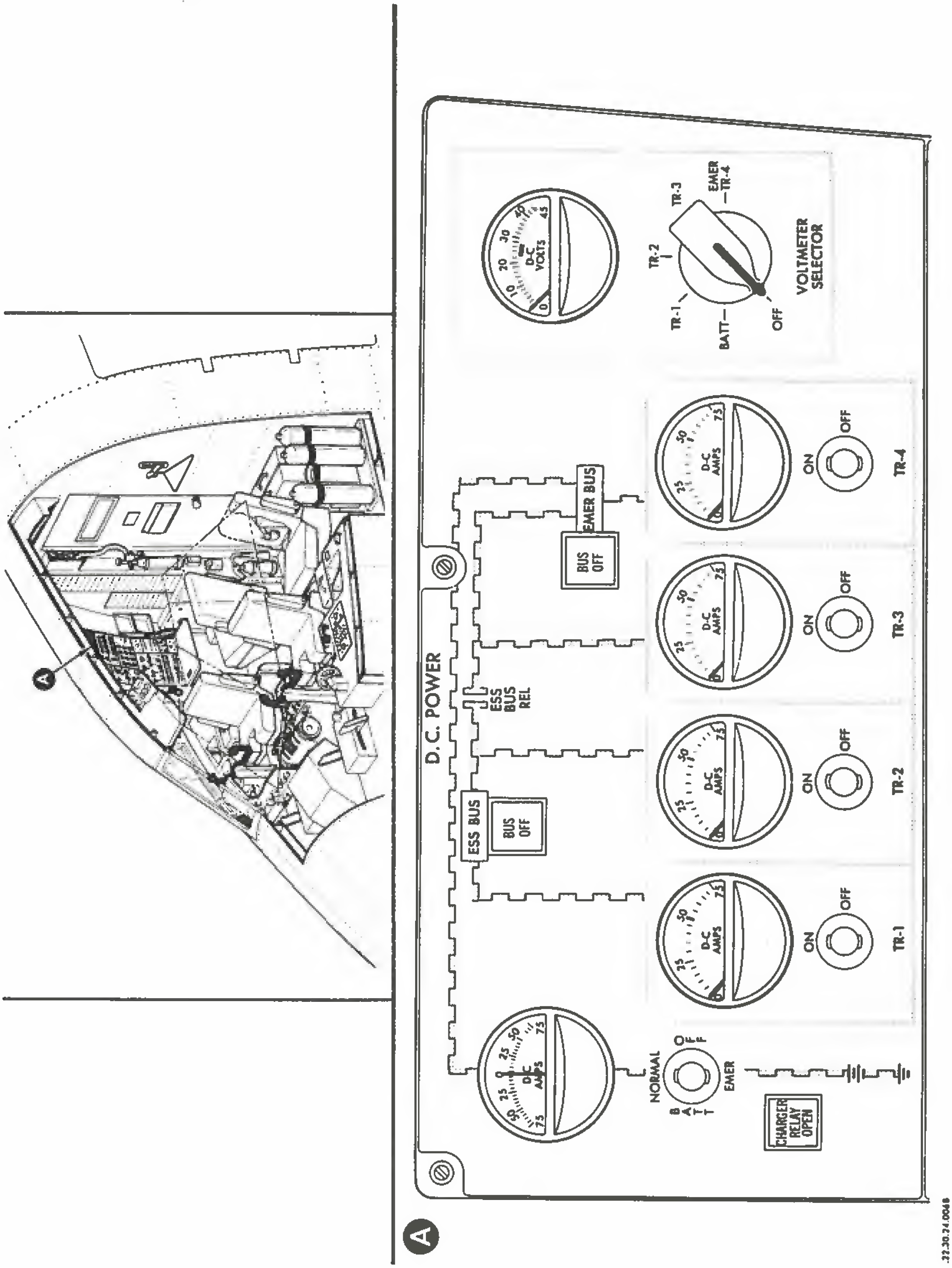
The essential dc bus, located in the dc junction box, supplies power to all essential dc loads and to the nonessential dc loads through current limiters and circuit breakers. The emergency dc bus is located in the dc junction box. It supplies power from the main T-R units or the battery to all the emergency circuits through current limiters and circuit breakers.

FLIGHT ENGINEER'S DC CONTROL PANEL

The dc electrical system is monitored and controlled through the instruments and switches located on the flight engineer's dc control panel (see Figure 13-10).

Main T-R Unit Ammeters

Four ammeters, one for each main T-R unit, are connected to the negative lead of each main T-R unit. The meters are external shunt-type, calibrated from 0 to 75 amperes.



Flight Engineer's DC Control Panel
Figure 13-10

Battery Ammeter

The battery ammeter is an external shunt-type, 75-0-75 amperes, connected in the negative leg of the battery. One half of the scale indicates battery discharge current with the battery switch in the EMER position. The other half of the scale indicates battery charging current with the battery switch in NORMAL position. The meter is located on the flight engineer's panel.

NOTE: The battery ammeter can be interpreted to indicate state of charge of the battery. If the charging current is low (approximately 0.5 amperes), the battery is charged. If the meter indicates a charging rate of around 20 amperes, it is discharged. The charger will charge a dead battery in about one hour, during which the ammeter reading will vary from an initial value of about 20 amperes to the final value of 0.5 amperes.

Voltmeter

A voltmeter and a six-position selector switch OFF, BATT, TR-1, TR-2, TR-3, EMER TR-4 are provided on the flight engineer's dc panel. During normal system operation, the switch may be placed in the various positions and the corresponding voltages monitored.

DC SYSTEM WARNING LIGHTS

Three amber lights, located on the engineer's dc panel, indicate the electrical connections in the dc system.

When the battery switch is in the NORMAL position, and the pilot's essential bus is energized, the CHARGER RELAY OPEN light will be off. In the EMER position, the light will illuminate.

The BUS OFF light in the 28-volt dc emergency bus circuit illuminates when the emergency dc bus is deenergized.

Section 14

AIR CONDITIONING AND PRESSURIZATION SYSTEMS

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AIR CONDITIONING AND PRESSURIZATION SYSTEMS

GENERAL AIR CONDITIONING AND PRESSURIZATION

The air conditioning and pressurization system provides passenger and crew comfort under all anticipated temperature conditions at airplane altitudes up to 41,000 feet. The pressurization equipment maintains sea level pressure in the cabin up to an airplane altitude of 21,300 feet and up to an 8000-foot cabin pressure at 41,000 feet. The air conditioning equipment supplies a complete change of air in the cabin every two and one-half minutes and in the crew compartment every minute. The inside temperature can be maintained between 18 degrees C and 33 degrees C (65 degrees F and 90 degrees F), dependent on passenger load, with outside temperatures -54 degrees C to 38 degrees C (-65 degrees F to 100 degrees F). (See Figures 14-1 and 14-2.)

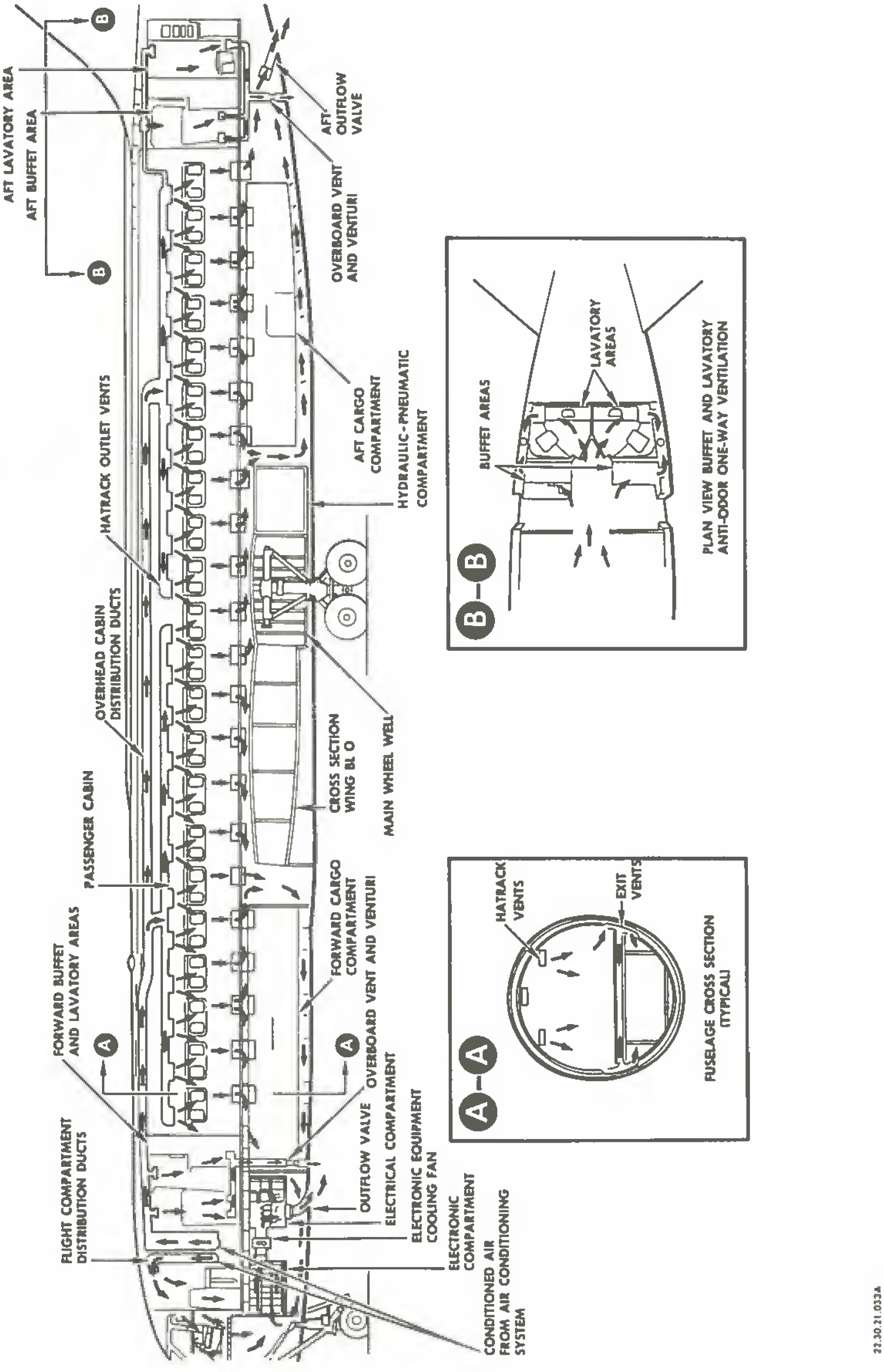
The cabin pressure and temperature are normally controlled automatically, but can be regulated manually at the discretion of the crew. The change of cabin pressure during rapid descents can be controlled automatically at any preset rate between 50 to 2000 feet per minute.

The airplane is equipped with two independent and parallel pressurization and air conditioning systems. One system supplies air to the flight deck, the other to the cabin (see Figure 14-3). Two turbocompressors, driven by engine bleed air, provide an ample supply of compressed air to maintain the required cabin-air density during high-altitude operations. Inflight heating is provided by the compressing action of the turbocompressors. Two electric heaters are available for heating when the airplane is on the ground. An air-to-air heat exchanger and a refrigeration unit in each system provide the cooling required to keep the cabin air at the selected temperature.

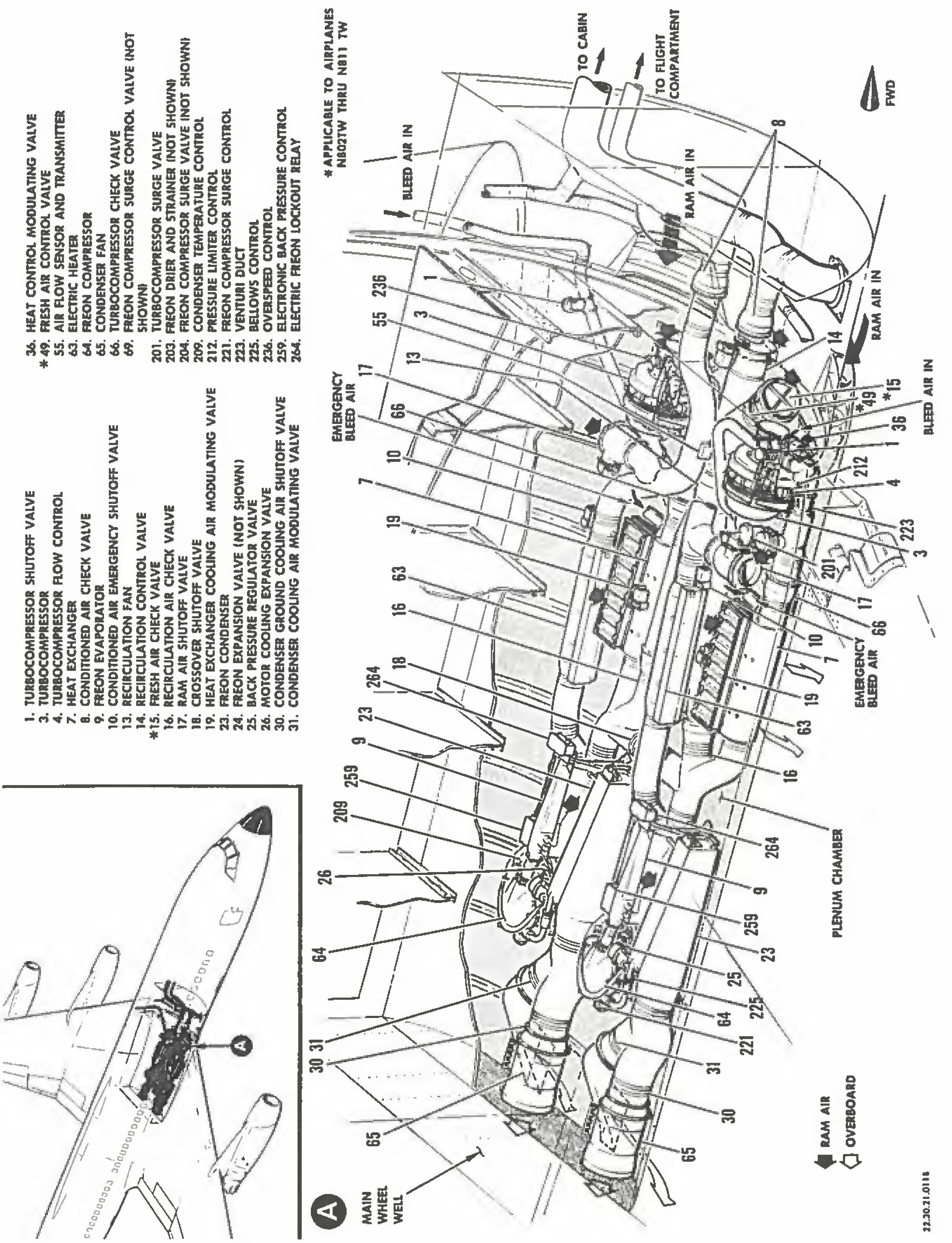
A recirculating fan can be used for recirculating the air through the cabin. A crossover duct provides a means of connecting the two systems if one turbocompressor should fail. If both turbocompressors should fail, the systems can be operated by bleed air from the engines. (See Section 2, EMERGENCY PROCEDURES.)

The gages and switches for monitoring and controlling the operation of the air conditioning and pressurization system are located on the flight engineer's panel. The cabin presusre altitude, the cabin temperature, and the rate of change of cabin pressure can be regulated automatically or controlled manually.

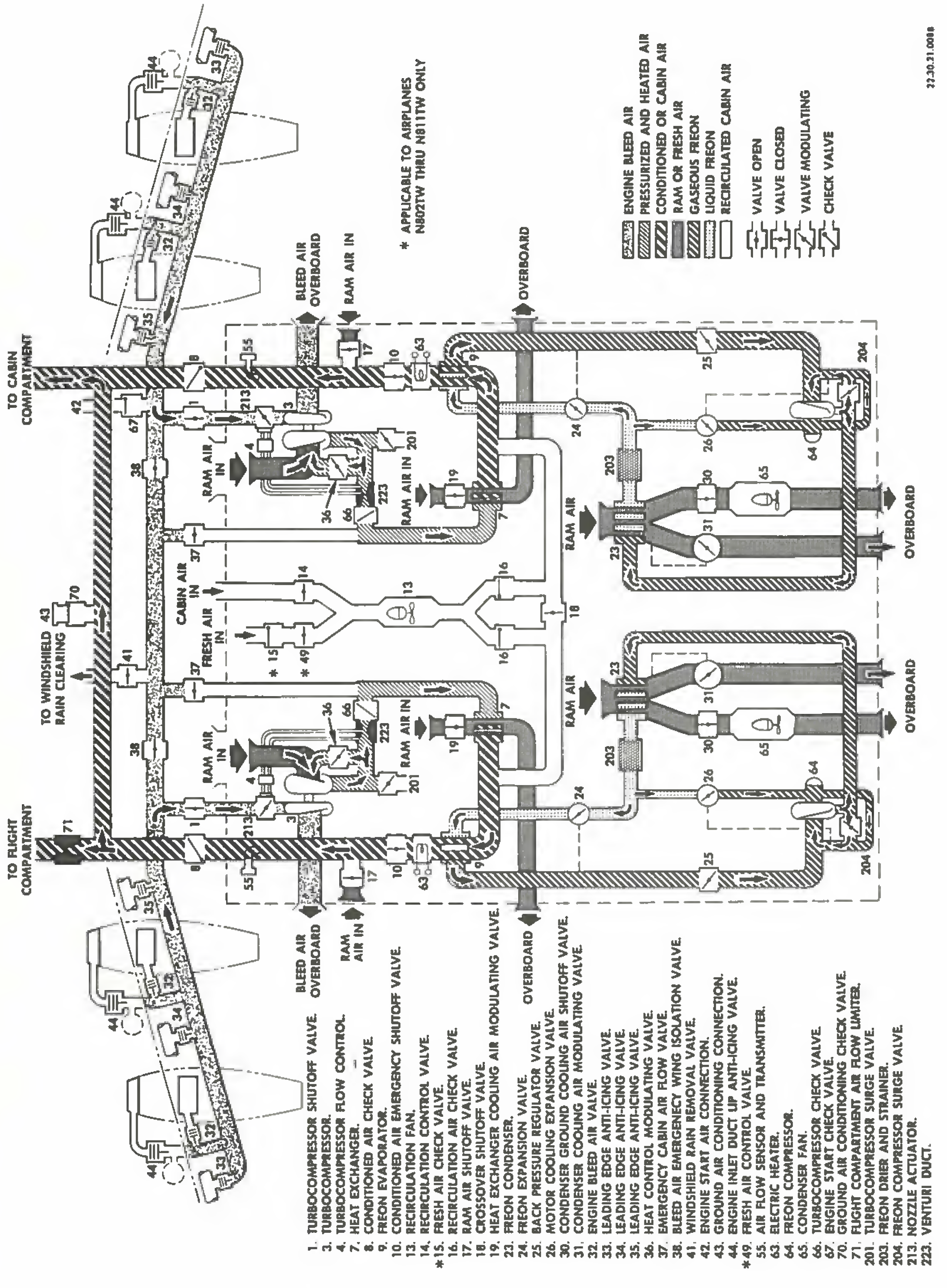
As the two air conditioning systems are identical, it will suffice to describe only one of them. Essentially, each system consists of a turbocompressor; a primary air-to-air heat exchanger; a refrigeration package; valves, regulators and switches for controlling the operation of the system; and the necessary air passages and ducts.



Air Conditioning
and Pressurization Airflow
Figure 14-1



Air Conditioning Package
Figure 14-2



Air Conditioning System Schematic
Figure 14-3

TURBOCOMPRESSOR UNIT

The turbocompressor supplies pressurized fresh air to the air conditioning system at the proper weight-flow and pressure to maintain the preselected cabin pressure. Each turbocompressor consists of a single-stage turbine and a centrifugal single-stage compressor mounted on a common shaft.

Turbine

The turbine is driven by bleed air from the main bleed-air manifold which receives air from the seventeenth compressor stage of each engine through a pressure regulator that reduces the pressure of the bleed air to 40 psig. After passing through the turbine, the bleed air is dumped overboard (see Figure 14-4).

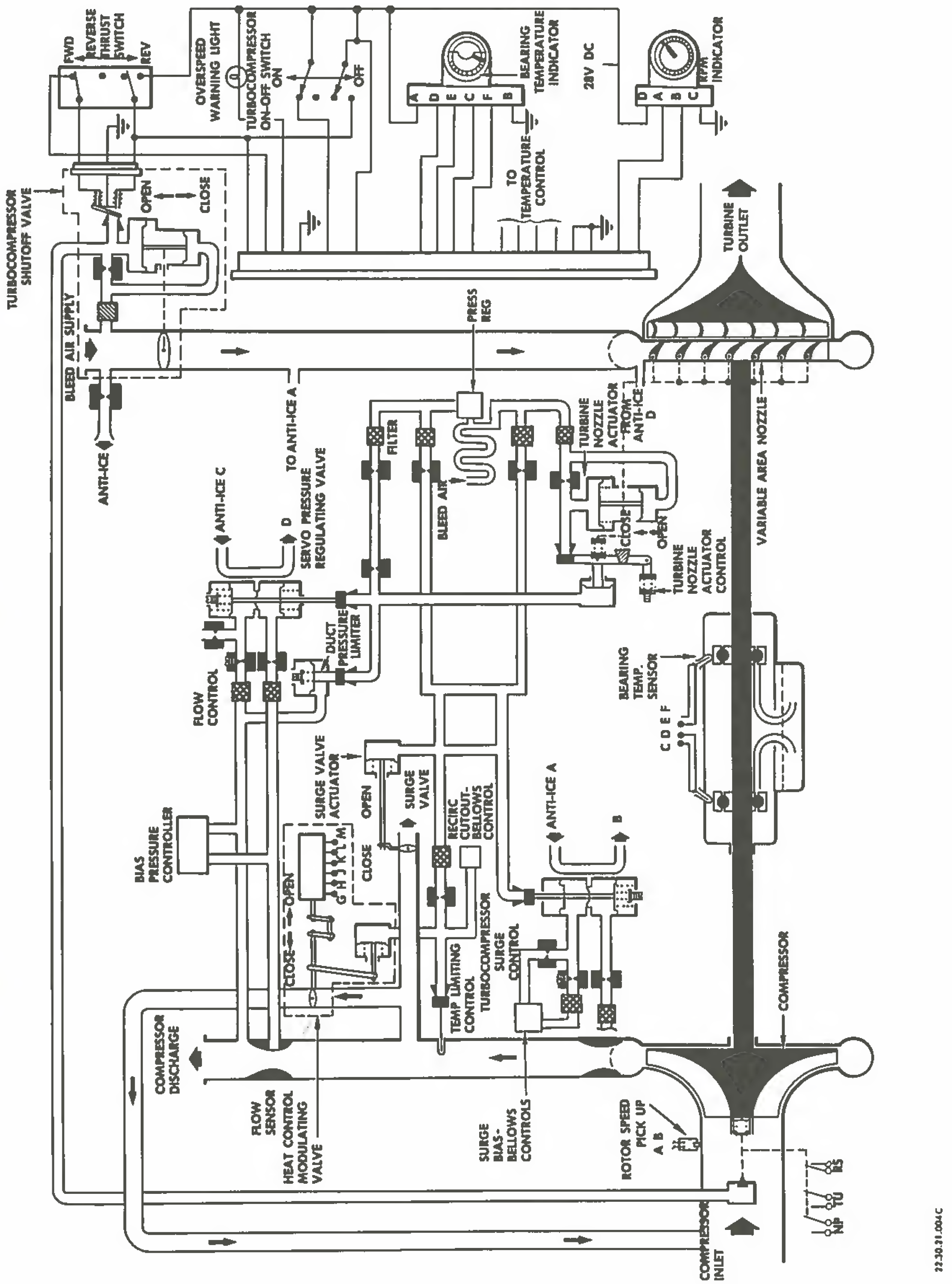
Bleed air enters the turbine through turbocompressor shutoff valve to a variable area nozzle controlled by a flow sensor at the compressor outlet. The sensor detects variations in airflow and automatically increases or decreases the nozzle opening to adjust the turbine speed accordingly. An rpm indicator, indicating turbine speed, is located on the flight engineer's panel. An overspeed control shuts off the supply of bleed air if the turbine speed increases to between 54,000 and 56,000 rpm. When the turbine overspeed cutout is tripped, the unit must be reset manually on the ground. The turbocompressor automatically shuts off during reverse thrust use, thus preventing the entry of exhaust gases into the cabin.

Compressor

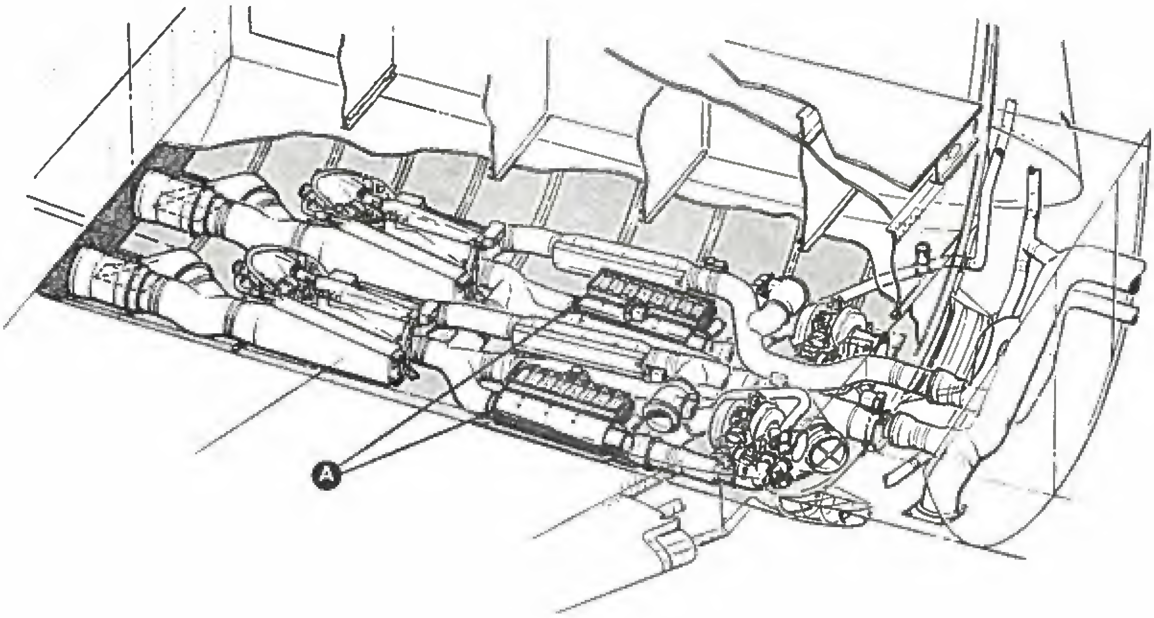
Air conditioning ram air enters a plenum chamber through two large air scoops beneath the fuselage near the wing leading edge and passes into the compressor inlet. As the air passes through the compressor it is compressed and heated. If additional heating is required, a portion of the compressed air is recirculated through the compressor. A heat control modulating valve, regulated by the temperature control sub-system in the air conditioning system, controls the percentage of the compressor discharge air to be recirculated. A surge valve, also located at the compressor outlet, diverts compressed air back to the plenum chamber when a decrease in airflow through the compressor is accompanied by a marked increase in outlet pressure. This condition would arise in the case of a blocked duct. A check valve prevents reverse flow through the compressor when it is shut down.

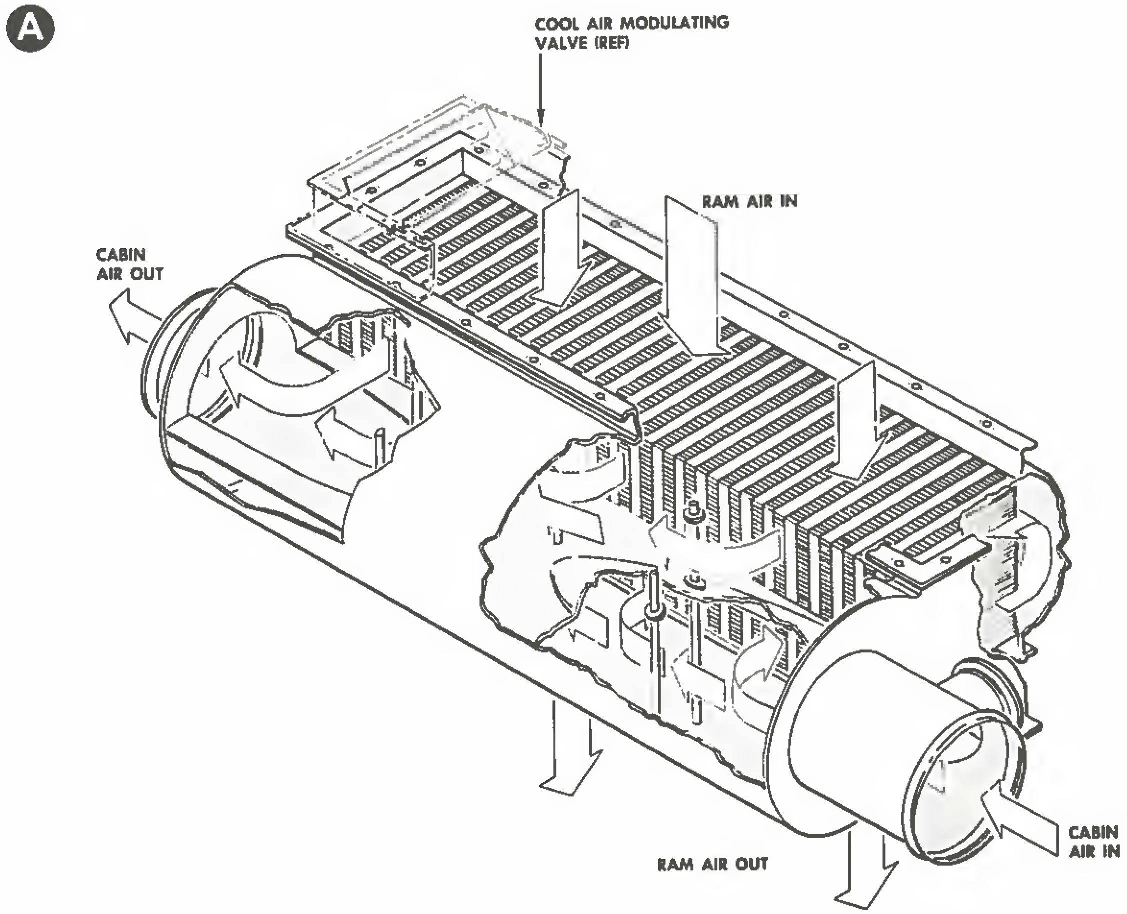
AIR TO AIR HEAT EXCHANGER

From the compressor, the heated compressed air passes through an air-to-air heat exchanger for primary cooling. The heat exchanger has a double-deck plate and fin core with vertical passages for ram air and horizontal passages for compressed cabin air (see Figure 14-5). The cabin air passes over and back through the two decks of the core. The ram air makes a single pass vertically through the core and absorbs heat from the plates and fins thus cooling the cabin air flowing through the horizontal passages.



Turbocompressor Operational Schematic
Figure 14-4





Heat Exchanger Unit
Figure 14-5

The ram air enters the heat exchanger through a cooling air modulating valve controlled by the temperature control sub-system of the air conditioning system. The amount of cooling varies from full capacity cooling to no cooling, depending on the position of the modulating valve.

FREON COOLING SYSTEM

Secondary cooling is provided by a closed vapor-cycle refrigeration unit located in the air conditioning compartment of the airplane. The unit consists of a compressor, condenser, evaporator, and associated control devices (see Figure 14-6). Freon 114, a nonpoisonous fluid which boils at 3.5 degrees C (38.4 F), is used as the refrigerant. Special oil is mixed with the Freon, about 1 part in 20, to lubricate the bearings and other internal moving parts. The system is designed for either automatic or manual control and will maintain stable operation when adjusted for any amount of cooling between full capacity and essentially no cooling.

A vapor-cycle refrigerator depends on the latent head of condensation and vaporization of a refrigerant to produce the desired drop in ventilating air temperature. The refrigerant, in a gaseous state, is passed through a compressor where it is compressed and heated. The compressed gas is then cooled and condensed in a heat exchanger that disposes of the latent heat released when the gas condenses. The refrigerant, now in the liquid state, flows through another heat exchanger where it changes back to gas. The heat needed to vaporize the liquid is extracted from the cabin air which the unit was designed to cool. From the evaporator, the gas is returned to the compressor and the cycle is repeated.

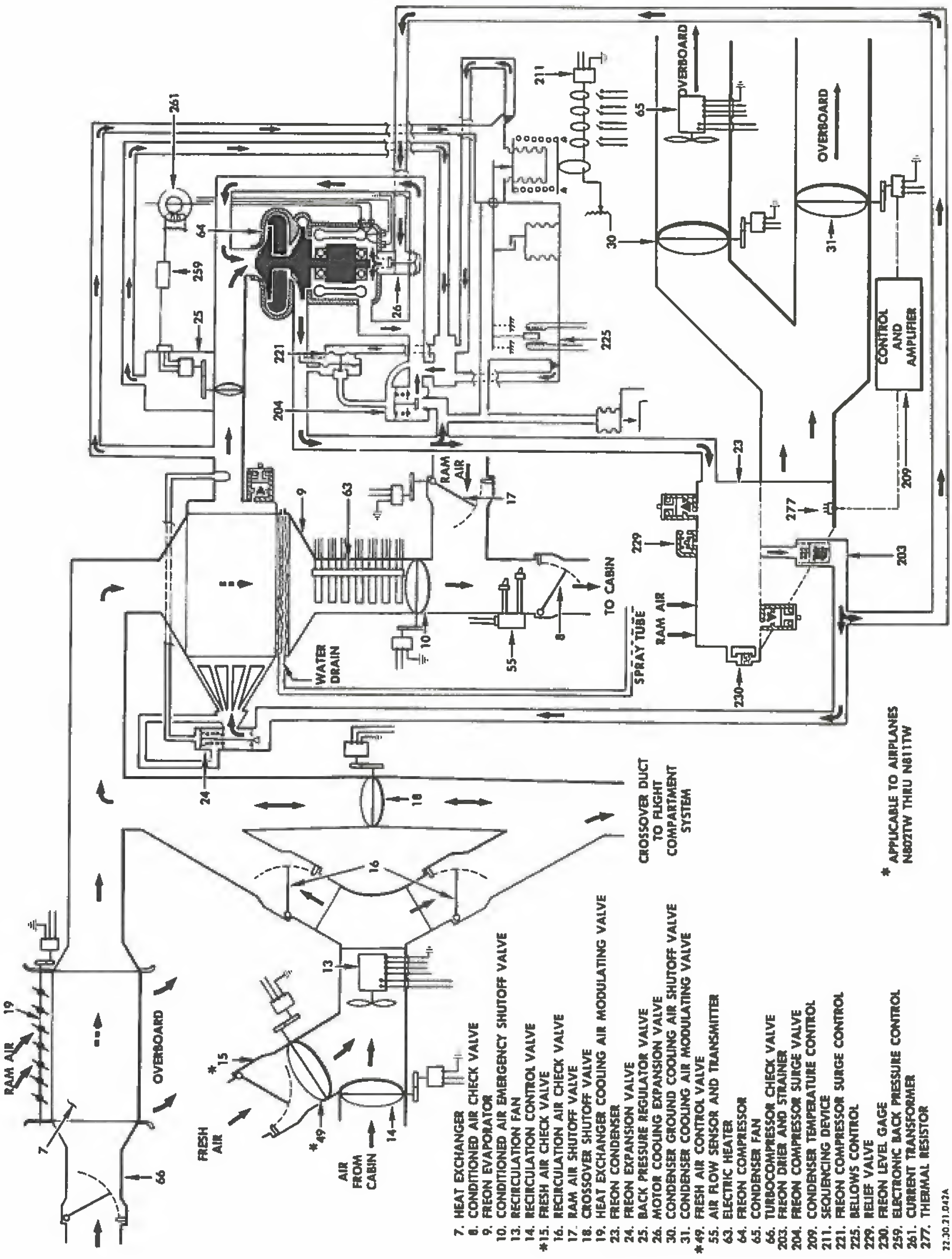
Freon Compressor

The Freon gas is compressed and heated in a two-stage centrifugal compressor with sweptback blades to permit a wide stable operating range for various compressor flows at nearly constant speed. The compressor is driven by a 3-phase, 400-cycle, 115/200-volt ac, squirrel-cage induction motor. The motor will develop about 25 hp and provide an 8-ton cooling capacity at a power factor of approximately 85 percent.

The motor is cooled and lubricated by circulating Freon through the bearings and passages in the motor housing. A back-pressure controller limits the operating current to 88 amperes per phase and a thermal protective device shuts down the motor if the temperature of the windings rises to 149 degrees C (300 degrees F).

Freon Condenser

The Freon condenser is a Freon-to-air type heat exchanger in which the hot compressed Freon gas is cooled and condensed to a liquid. Cooling is provided during most flight conditions by allowing ram air to pass through the condenser. For slow flight and ground cooling, plenum air is drawn through the condenser



Freon Refrigeration System Functional Diagram
Figure 14-6

by an electric fan. Sufficient condenser cooling is provided to change the Freon gas to a liquid and subcool it to keep the liquid from flashing into a gas when it leaves the condenser.

The cooling air leaves the condenser through a Y duct. In flight, the cooling air discharges through the condenser cooling air modulating valve and is dumped overboard. On the ground, the cooling air leaves the condenser through the ground cooling air shutoff valve, passes through the fan, and empties into the main landing gear well. The modulating valves are regulated automatically to maintain the correct Freon temperature.

A liquid level gage provides a visual indication of the amount of Freon in the condenser. For an accurate check, the reading should be taken following operation and the condenser should be as cold as possible. A spring-loaded relief valve on the top of the condenser will open if the pressure in the condenser reaches 180 psia. This pressure is well above the normal operating range.

A Freon drier and a strainer unit with a replaceable cartridge is located at the condenser outlet. The cartridge contains a drying agent surrounded by a filter screen. All Freon leaving the condenser is strained by the filter screen to prevent the circulation of foreign particles through the system. A portion of the Freon also comes in contact with the desiccant which removes any moisture and prevents corrosion within the system.

Freon Evaporator

From the condenser, the cold liquid Freon flows through a thermostatic expansion valve to the evaporator where it is completely vaporized before it returns to the compressor to complete the vapor cycle. The heat to "boil" the liquid Freon is extracted from the incoming cabin air.

The evaporator is a Freon-to-air heat exchanger made up of an outer shell and two heaters connected by a core containing 48 tubes. The Freon flows through the tubes while the cabin air from the turbocompressor passes through the spaces between the tubes and the outer shell. As the cabin air is cooled to the dew point, the moisture in the air condenses on the core tubes. A fine mesh screen separates the small droplets from the cabin air. The water drains into a water collector and is dumped overboard.

The thermostatic valve in the Freon line from the condenser regulates the flow of Freon into the evaporator and the degree of superheat at the evaporator outlet. The operation of this valve is controlled by temperature and a pressure sensing element in the outlet header of the evaporator. The flow of Freon into the evaporator is restricted to a rate that will assure complete vaporization and additional heating of the vapor before it leaves the evaporator. This valve also prevents liquid Freon from entering the evaporator when the Freon package is shut down.

The boiling temperature of the liquid Freon depends on the vapor pressure within the evaporator. This temperature varies from -1.1 degrees C (30 degrees F) at a pressure of 12.25 psia to 18.3 degrees C (65 degrees F) when the pressure is increased to 25 psia. The cooling provided by the evaporator is regulated by controlling the vapor pressure within the evaporator through the action of a back-pressure valve located in the Freon line between the evaporator and the compressor. The action of this valve is controlled automatically by the cabin temperature control system.

Motor Cooling Loop

A second Freon loop is provided to cool the motor and compressor. Liquid Freon from the condenser outlet passes through a thermostatic expansion valve into the section of the compressor housing enclosing the electric motor. From there it empties into the compressor inlet. This valve is controlled by a unit which senses the temperature of the Freon after it leaves the motor housing. In this loop, the liquid Freon is vaporized by heat extracted from the motor much as it is vaporized in the evaporator by heat extracted from the cabin air.

Surge Control

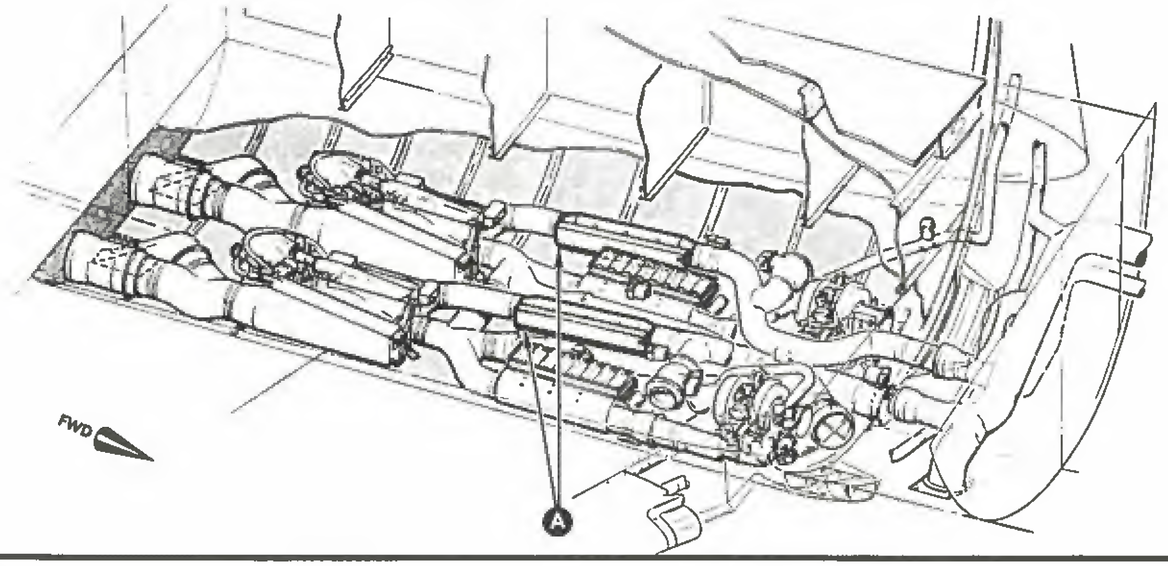
A third Freon loop connecting the compressor outlet with the compressor inlet controls pressure surging in the compressor. Freon flow through this loop is controlled by a surge valve. In case of limited weight-flow accompanied by an excessive pressure differential through the compressor, the surge valve will open and some of the Freon will be recirculated. The increase in turbine temperature due to recirculating the Freon causes the motor-cooling thermostatic valve to allow more Freon to circulate through the motor cooling loop. Thus, during times of Freon recirculation through the surge valve, the compressor motor is overcooled to limit the Freon temperature at the compressor outlet.

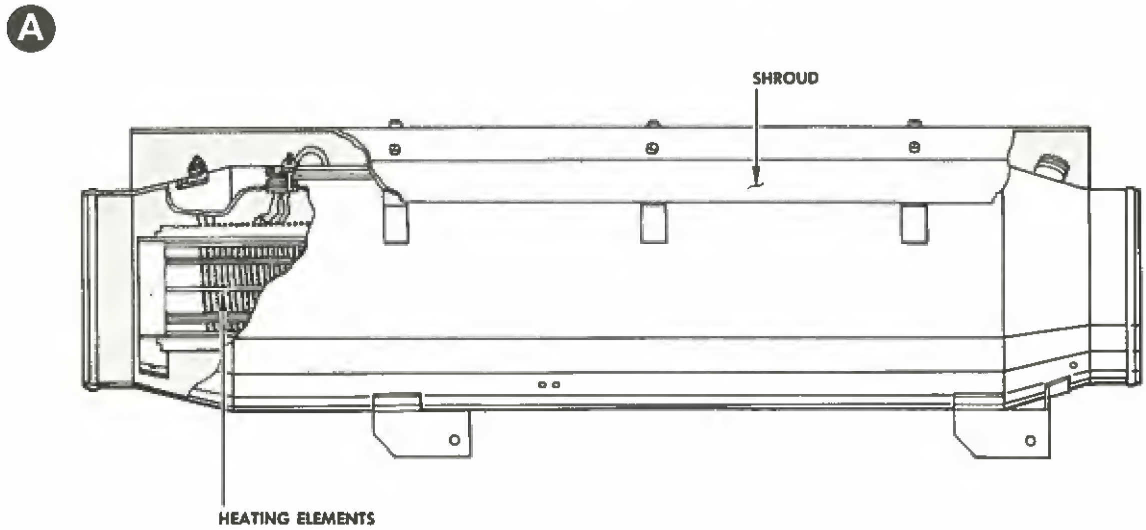
ELECTRIC HEATER

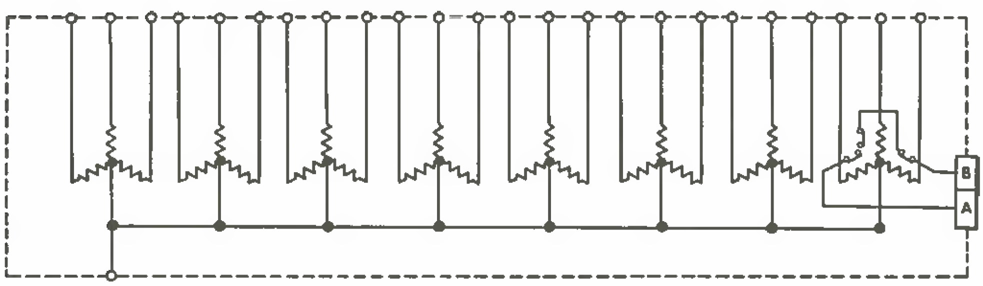
From the evaporator, the cabin air passes through an electric heater containing seven heating elements and then through an emergency cabin air shutoff valve. These elements are step sequenced so that the heat output can be increased in seven increments. The electric heater is operated only when the airplane is on the ground. The recirculating blower is on, and the emergency shutoff valve is open. The heater is controlled automatically by the cabin temperature control system (see Figure 14-7).

CABIN AIR RECIRCULATING SYSTEM

The recirculating system is used primarily to provide cabin ventilation when the airplane is connected to a ground cart or when taxiing on a hot day. In flight, the system can be used to improve ventilation when one of the turbocompressors is shut down. The system is connected to the flight deck and cabin air conditioning systems through the evaporator crossover duct and one-way check valves.







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Electric Heater Unit
Figure 14-7

The recirculating system consists essentially of a recirculating blower driven by a 400-cycle, 115/200-volt ac electric motor; a modulating cabin air control valve; a modulating fresh air valve; and a fresh air check valve. The blower is controlled through an ON-OFF toggle switch on the flight engineer's panel.

ALTERNATE PRESSURIZATION

Bleed air can be used for alternate cabin pressurization. Two bleed air lines connect the bleed air manifold with the cabin and flight deck air conditioning systems downstream from the turbocompressors. Two toggle switches on the flight engineer's panel control the bleed air control and shutoff valve in each bleed air line. Since the bleed air passes through the primary heat exchangers and the Freon evaporators, its temperature is regulated by the cabin temperature control system (see Figure 14-8).

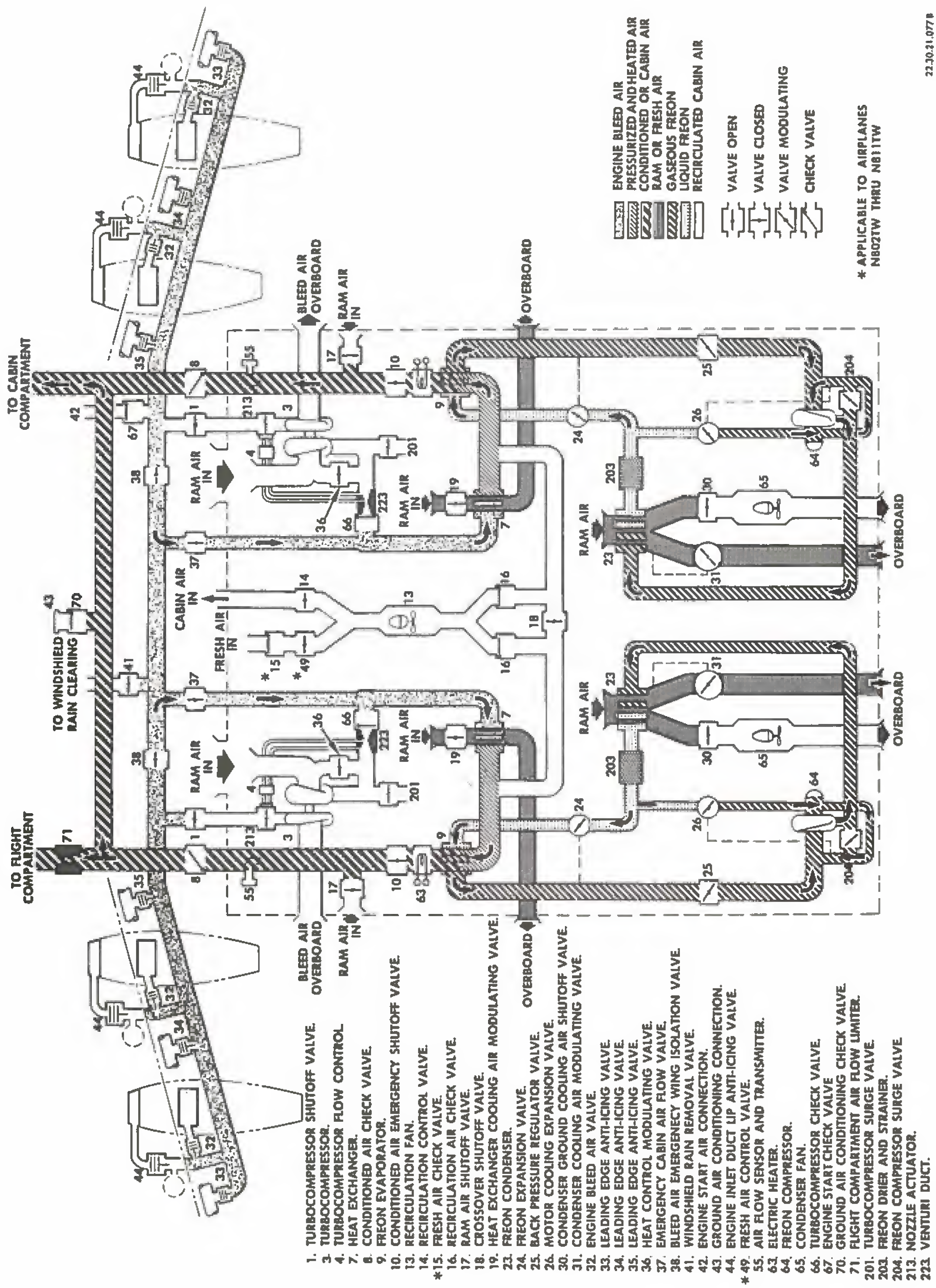
CABIN PRESSURE CONTROL

Two cabin air outflow valves maintain the selected pressure schedule and provide the required airflow for each particular altitude. Either outflow valve is capable of maintaining cabin pressure within the prescribed limits. Normally, these valves are regulated automatically, but they can be adjusted manually by two 4-position switches on the flight engineer's panel.

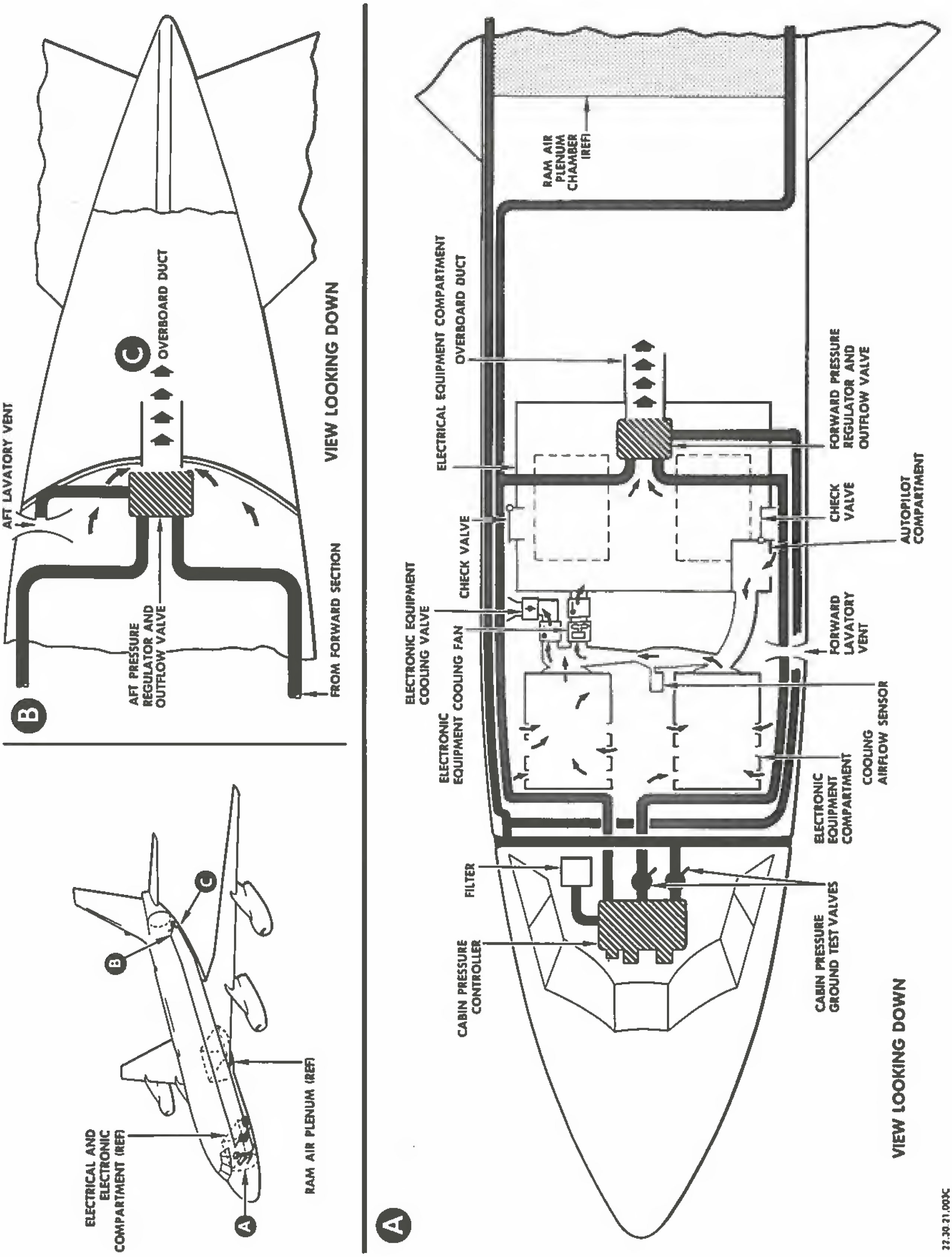
The outflow valves are set to limit the differential between the cabin pressure and that of the surrounding atmosphere to 8.3 psi. If both valves should fail, a safety feature in each valve will limit the pressure differential to 8.6 psi. If the valves should malfunction, they will close automatically.

One of the outflow valves is located in the aft pressure area of the cabin and discharges directly overboard. The forward outflow valve is located in a plenum chamber housing the electrical equipment (see Figure 14-9). Air from the flight deck and the forward area of the cabin is channelled into the electrical chamber through a check valve in the electronic compartment. From the electrical chamber, the air is discharged overboard through the outflow valve.

An electronic equipment cooling fan and a flow sensor are located in the passage connecting the electronic and electrical compartments. A tee connection on the duct between the electronic rack and the cooling fan connects to a venturi which discharges overboard. During pressurized flight condition, the air discharges overboard through the flow limiting venturi. An electrically actuated shutoff valve is used upstream of the venturi to prevent loss of cabin pressurization if one turbocompressor is not operating. The flow sensor illuminates a warning light on the flight engineer's panel if the airflow through the electronic racks is not sufficient to cool the equipment. The cooling fan is turned on automatically by the oleo scissors switch when the airplane is on the ground, and it can be turned on manually in flight should the electronic cooling valve fail in the closed position and the low airflow warning light illuminates.



Emergency Pressurization System Schematic
Figure 14-8



Pressurization System Control Valves
Figure 14-9

If the aft outflow valve should fail in the closed position, an inward-opening check valve allows more air to enter the electrical chamber directly from the cabin. If the forward outflow valve should fail in the closed position, an outward-opening relief valve allows air from the cooling fan to discharge into the cabin.

VALVES

The control and operation of the air conditioning and pressurization system depends primarily on the functioning of various valves located throughout the system. Check valves are placed at strategic points in the system to prevent reverse flow; modulating valves regulate the rate of flow through the passages in which they are located; and shutoff valves are utilized to start and stop certain operations. Some of the modulating valves also serve as shutoff valves.

Check Valves

A check valve is located between each turbocompressor and corresponding emergency bleed air inlet to prevent reverse flow through the turbocompressors when the bleed air is turned on. A second pair of check valves is located in the ducts leading to the flight deck and the cabin, upstream from the crossover passage connecting the two air conditioning systems. When one of the systems has been shut down, the corresponding valves will close and keep the conditioned air from entering the inoperative system. This also prevents loss of pressurization should a duct break occur in the plenum area. Another pair of check valves isolates the recirculating blower from the two air conditioning systems when the blower is not operating.

Another check valve is located in the ground air conditioning connection to close this opening except when conditioned air is being furnished by a ground air conditioning unit.

Modulating Valves

The speed of each turbocompressor is regulated by a variable-area nozzle functioning as a modulating valve to control the flow of bleed air through the turbine. The recirculation of compressed air through the compressors is regulated by the cabin heat control modulating valves. Two modulating valves regulate the flow of cooling ram air through the primary heat exchangers and an additional pair of modulating valves controls the flow of ram air through the Freon condensers.

The air supply for the recirculating blower is controlled by a shutoff valve in the cabin air recirculating duct. This valve also functions as a shutoff valve to close the passage when not in use. The cabin air outflow valves modulate the rate at which the cabin air is released to the atmosphere and close the outflow openings when conditions warrant.

Four modulating valves in each Freon package regulate the flow of Freon through the three Freon loops. The Freon back pressure valve controls the cooling output of the unit; one expansion valve controls the flow of liquid Freon to the evaporator, another expansion valve controls the flow of liquid Freon through the compressor bearings for cooling and lubrication, and a fourth modulating valve regulates the recirculation of Freon gas through the compressor.

Shutoff Valves

Most of the air conditioning and pressurization equipment is turned on and off by shutoff valves. The turbocompressors are turned on by opening the turbocompressor bleed air shutoff valves. Two shutoff valves in the emergency bleed air ducts supply bleed air to the air conditioning systems when both turbocompressors are out. An emergency shutoff valve in each system can be used to close either the duct leading to the cabin or the one leading to the flight deck. Two motorized ram air check valves, one in each ram air inlet, can be opened to admit ram air directly to the flight deck and the cabin. These valves cannot be opened when the cabin pressure is higher than that of the ram air.

SYSTEM OPERATION

Ram air from the plenum chamber is compressed and heated by the compressors. The cabin heat control modulating valves regulate the amount of air that will be recirculated through the compressors for additional heating. The compressed air then passes through check valves and enters the primary heat exchangers where it is cooled, as required, by ram air from the plenum chamber. The amount of cooling is regulated by modulating valves that control the flow of cooling ram air through the heat exchangers.

The compressed air then passes through the Freon evaporators where its temperature may be further reduced by the vaporizing action of the Freon. From the evaporator, the air flows through the electric heaters that can be used to heat the incoming air while the airplane is on the ground.

The conditioned air flows past the cabin in-flow sensors and transmitters that provide an indication of turbocompressor, or fan, weight flow. It then flows past the openings of the crossover duct connecting the two air conditioning systems and through the flow limiters that provide the flight compartment with about 10 percent of the output of the two systems.

TURBOCOMPRESSOR OPERATION

The following operating tolerances may be used to aid in proper operation of the turbocompressors. The normal operating temperature for turbocompressor bearing operation is 150 to 190°F. Bearing temperature is such that any increase in turbo rpm may be accompanied by gradual increase in bearing temperature and such increase should be considered normal provided the maximum temperature of 250°F is not exceeded. Any abrupt increases in bearing temperature, not accompanied by an increase in rpm should prompt shutdown of the unit before reaching the maximum operating temperature.

The rpm spread between turbocompressors should not exceed 4000 rpm. The following may be used as a guide for turbocompressor speed and air flow value under various operating conditions:

Condition	Altitude	RPM (±2000)	Airflow (±10 PPM)
Climb	Sea Level	23,000	92 Pounds
Cruise	15,000	30,000	80
Cruise	20,000	33,000	75
Cruise	25,000	36,000	70
Cruise	35,000	42,000	65
Cruise	40,000	47,000	55
Descent	40,000	45,000	53
Descent	35,000	42,000	67
Descent	20,000	27,000	77
Descent	10,000	28,000	80

The accuracy of the airflow indicating system on each side may be checked inflight or on the ground as follows:

In-Flight

Compare airflow readings between right and left sides by actuating appropriate turbocompressor control switch (ON - OFF) and crossover valve by using the Freon compressor selector switch.

On Ground

a. Place cabin temperature control switch in AUTO or MANUAL.
b. Place recirculating blower switch ON. Observe that airflow gage indicates approximately 85 ppm flow through each side of the system.
c. Place recirculating blower switch OFF. Observe that airflow gage indicates no flow through either side of the system.

The recirculating blower is used for air conditioning on the ground and to improve ventilation in flight when one of the turbocompressors has been shut down. The outflow valves allow the air to escape from the cabin at a rate that will maintain the scheduled cabin pressure.

AIR CONDITIONING AND PRESSURIZATION SYSTEM CONTROLS

The instruments and controls for monitoring and regulating the air conditioning and pressurization systems are located on the flight engineer's panel (see Figure 14-10). Four engine bleed-air shutoff switches are located on the pilots' overhead switch panel.

Instruments

Two COMPR RPM INDICATORS indicate the speed of each turbocompressor. Two COMP BEARING TEMP gages indicate the temperature of the turbocompressor bearings. A dual AIR FLOW gage registers the air flow from each air conditioning unit in lbs/min. An adjacent gage indicates CABIN temperature.

A CABIN ALTIMETER and CABIN DIFFERENTIAL PRESSURE indicator, a cabin CLIMB indicator and a CABIN PRESSURE CONTROL are located in the lower half of the control panel. The CABIN PRESSURE CONTROL has a large dial indicating pressure altitude from -1 to 10,000 feet and two cutouts: one exposing a sub-scale indicating barometric pressure in inches or mercury, the other showing a sub-scale indicating the maximum airplane altitude corresponding to the cabin altitude at which the hand on the large scale has been set (see Figure 14-11).

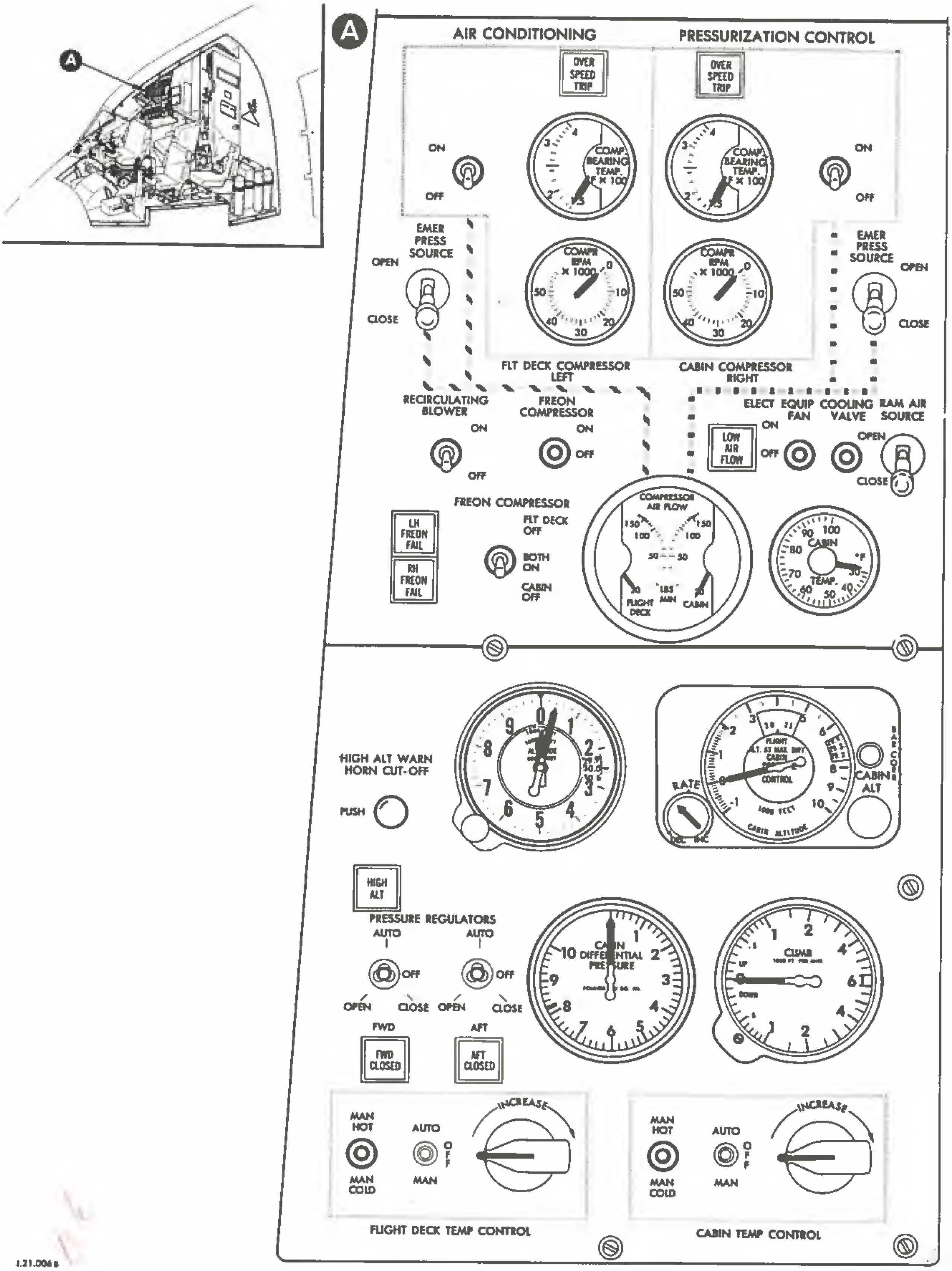
The CABIN PRESSURE CONTROL is equipped with three control knobs. The knob at the upper right controls the barometric sub-scale. The knob at the lower right turns the hand on the large dial to the selected cabin altitude. The knob at the lower left regulates the rate of change of cabin pressure. At takeoff and prior to landing, the large hand is adjusted to correspond to field altitude and the barometric scale is set at the reported airport pressure. Enroute, the hand on the large dial is set at a value which will provide passenger comfort without maintaining the cabin pressure any higher than necessary. When the airplane lands, the outflow valves are opened automatically, and the cabin pressure increases or decreases to match that of the outside air.

Air Conditioning and Pressurization System Control Switches

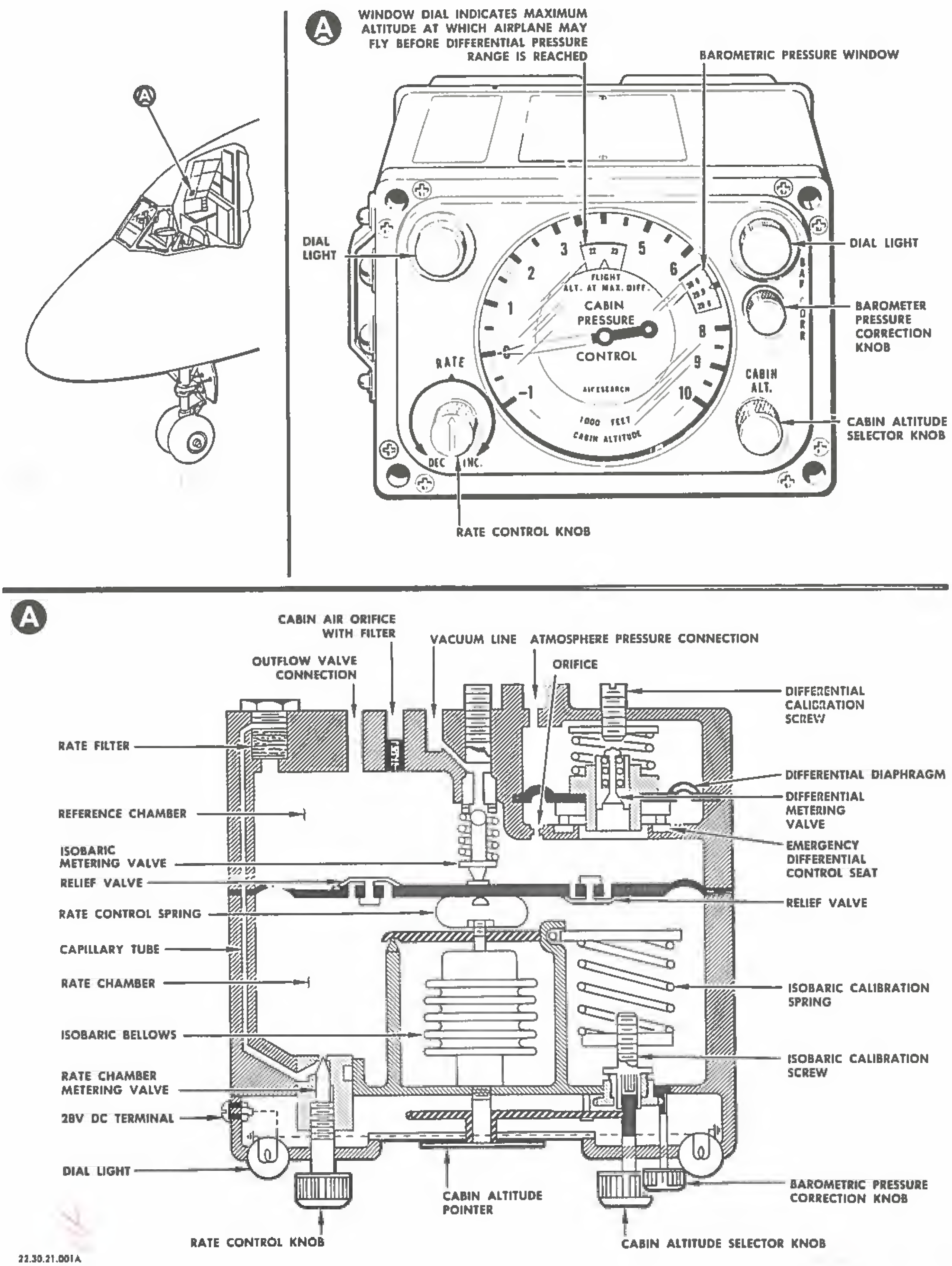
Two turbocompressor ON-OFF switches control the operation of the flight deck and cabin turbocompressors.

Two locking type EMER PRESS SOURCE switches OPEN or CLOSE the bleed air valves for emergency pressurization.

The RECIRCULATING BLOWER is controlled by a single ON-OFF toggle switch.



Air Conditioning and Pressurization
System Control Panel
Figure 14-10



Cabin Pressure Controller Unit
Figure 14-11

The FREON COMPRESSOR ON-OFF master switch controls the Freon compressors as selected by the three-positioned FREON COMPRESSOR selector switch. This switch placarded FLT DECK OFF-BOTH ON-CABIN OFF, is used for manual selection of the desired Freon compressor operational mode.

A RAM AIR SOURCE locking type toggle switch is used to OPEN or CLOSE the ram air valve.

The ELECT EQUIP COOLING, FAN is controlled by an ON-OFF toggle switch. The cooling VALVE is controlled by an OPEN-CLOSE toggle switch.

Two PRESSURE REGULATOR switches placarded AUTO-OPEN-CLOSE-OFF provide for manual operation of the two outflow valves. In the AUTO position, cabin pressure is regulated by the cabin pressure controller. The OPEN and CLOSE positions are used when operated under manual control to "inch" the outflow valves toward the open or closed position.

Four toggle switches and two thermostatic controls are located along the bottom of the control panel. The two AUTO-OFF-MAN three position switches are used to select the operating mode of the flight deck or cabin air conditioning systems. Two MAN HOT-MAN COLD momentary contract type switches are used to "inch" the temperature setting up or down when operating it in manual mode. Two thermostatic controls are used to regulate the temperature controls of the two systems when operating in the automatic mode.

<u>Air Conditioning and Pressurization System Warning Lights</u>

Eight air conditioning and pressurization system warning lights are located on the flight engineer's panel. Two amber OVERSPEED TRIP warning lights connect to the overspeed cutout valves of the two turbocompressors. If one of the turbocompressors should overspeed and trip the cutout valve, the corresponding light will come on.

Two amber warning lights, LH FREON FAIL and RH FREON FAIL, connect to overpressure switches and the overheat trip of the two Freon compressor motors. If one of the units closes down, the corresponding light will come on.

An amber FWD CLOSE and an amber AFT CLOSE light, adjacent to the pressure regulator switches, provide a warning if one of the cabin outflow valves goes to the closed position. A single red HIGH ALT light illuminates when the pressure altitude in the cabin exceeds 10,000 feet. A warning horn sounds simultaneously with the light. A HIGH ALT WARN HORN CUT-OFF switch, placarded PUSH, is provided.

An amber LOW AIR FLOW light adjacent to the airflow indicator will illuminate if the airflow for the electronic equipment is too low to provide adequate cooling.

ENGINE BLEED AIR SWITCHES

Four two-position OPEN-CLOSE switches on the pilot's overhead switch panel control the passage of bleed air from each engine to the bleed air manifold.

CLOSED indication lights illuminate when the corresponding valves are closed, and a HIGH DUCT PRESS light illuminates when the bleed air pressure in the manifold increases to approximately 50 psig. The malfunctioning pressure regulator valve can be identified by closing one valve at a time until the light goes out.

AUTOMATIC TEMPERATURE CONTROL

During normal flight and ground operation, the cabin temperature is controlled automatically by the temperature control system. This system consists of the cabin temperature control, a sequencing device, temperature selectors, and two thermal resistors that sense the cabin discharge and inlet duct air temperatures. The system maintains the cabin temperature in accordance with the setting of the temperature selector and limits the inlet duct temperature to 71 degrees C (160 degrees F). In response to signals from the thermal resistors and the temperature selector, the cabin temperature control positions the sequencing device which, in turn, schedules the operation of the heating and cooling components of the system.

During flight, the cabin temperature is controlled by the cabin heat control modulating valves, the primary heat exchanger modulating valves, and the Freon refrigeration units. On the ground, the temperature is regulated by the recirculating blower, the cabin air electric heaters, and the Freon refrigeration units. These components and systems supply the proper amount of heating or cooling to maintain the temperature of the cabin air at the selected level.

Inflight Scheduling

During flight, the recirculating valve and the electric heaters are not controlled by the temperature control system. The valve is closed and the heaters are off. Compressed air is supplied to the cabin and flight compartment by the two turbocompressors. Additional heating is provided, at altitudes below 12,000 feet, by the heat control modulating valves recirculating part of the pressurized air through the compressors. Primary cooling of the pressurized air is provided by the primary heat exchangers through the action of the cool air modulating valves. Secondary cooling is provided by the refrigeration units through the operation of the Freon compressor turbine control valves.

When maximum cooling is scheduled, the cabin heat control modulating valves are closed and the heat exchanger air modulating valves and the Freon compressor back pressure valves are open. The incoming air passes through the compressors only once and receives maximum cooling in the heat exchangers and Freon evaporators.

When less cooling is required, the sequencing device slowly closes the Freon compressor back pressure valves reducing the cooling capacity of the refrigeration units. When the Freon compressor back pressure valves are closed all

the way, the sequencing device will shut down the Freon compressors. If still less cooling is required after the refrigeration units have been shut down, the sequencing device slowly closes the heat exchanger cool air modulating valves reducing the cooling effect of the heat exchangers.

If the air is too cool in the cabin with both the heat exchangers and the refrigeration units inoperative, the sequencing device slowly opens the cabin heat control modulating valves to recirculate some of the pressurized air through the compressors. These valves are modulated by the temperature system to limit the cabin duct temperature to 71 degrees C (160 degrees F). (See Figures 14-12, 14-13 and 14-14.)

Ground Scheduling

During ground operation, the modulating valves controlling the turbocompressors and primary heat exchangers are not subject to control by the temperature control system. Various degrees of cooling and heating are available through regulating the cooling action of the refrigeration units by means of the Freon compressor back pressure valves, by adjusting the output of the electric heaters, and by recirculation of cabin air by the recirculating fan. When maximum cabin cooling is scheduled, the cabin air electric heaters are off, and the cabin air recirculating valve and the Freon compressor back pressure valves are wide open.

Cabin air is drawn through the recirculating valve by the recirculating fan and is ducted through the Freon evaporators for maximum cooling. The cooled air is then ducted to the cabin and flight deck through the electric heaters which are off when cooling is scheduled.

If still less cooling is needed, the sequencing device will shut down the Freon compressors if the cabin air does not need to be cooled.

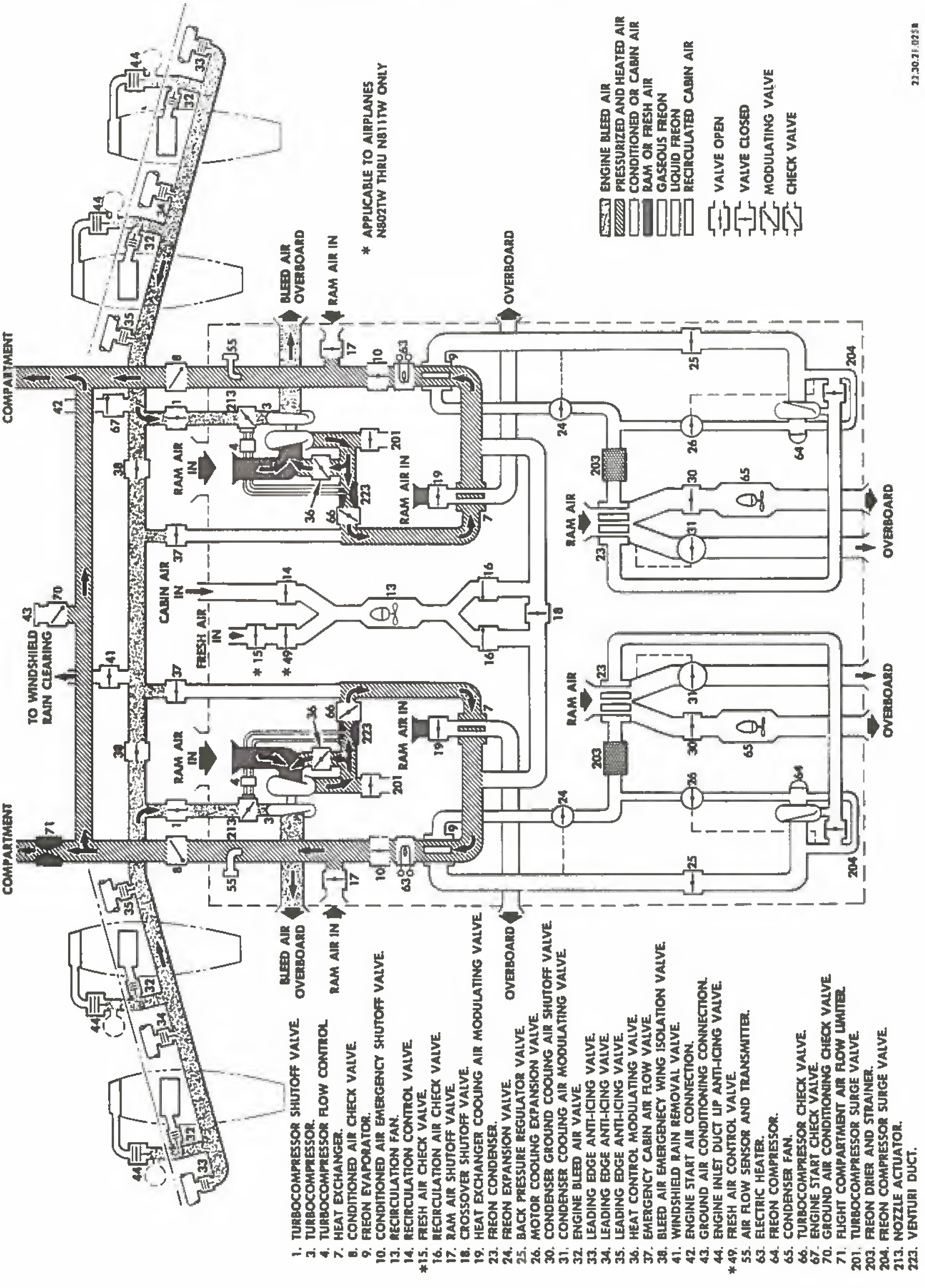
If heated cabin air is required, the sequencing device activates the cabin air electric heaters. The heating elements in the heater are energized in a series of stages as needed to meet the heating demand. Full heating is obtained with all the heating elements energized, the fresh air valve closed, and the cabin air recirculating valve open (see Figures 14-15, 14-16, 14-17 and 14-18).

POWER SOURCES

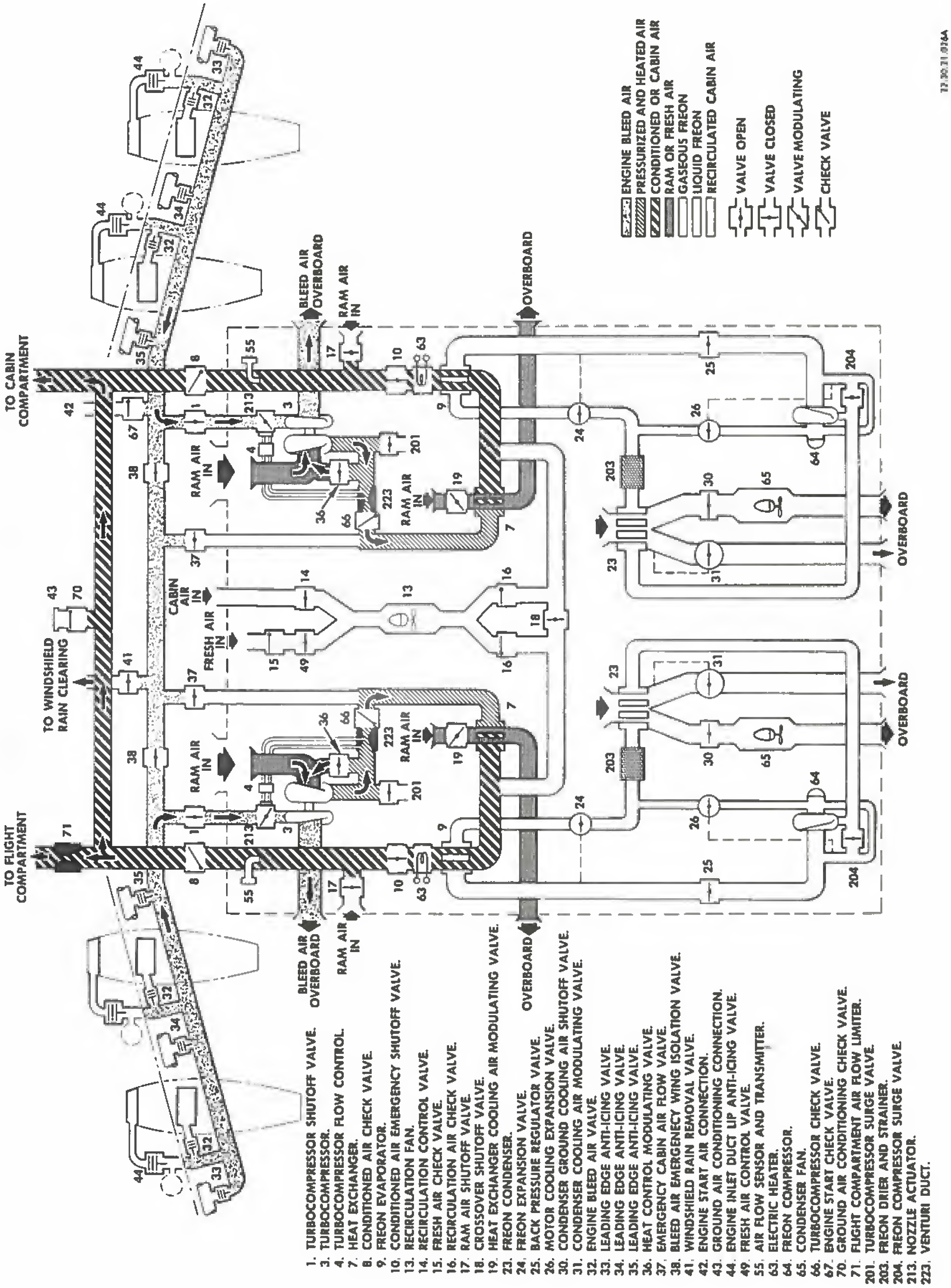
Power to operate the air conditioning and pressurization system is furnished by bleed air from the four turbojet engines and by the ac and dc electrical systems. For detailed electrical information, consult the WIRING DIAGRAM MANUAL.

Engine Bleed Air

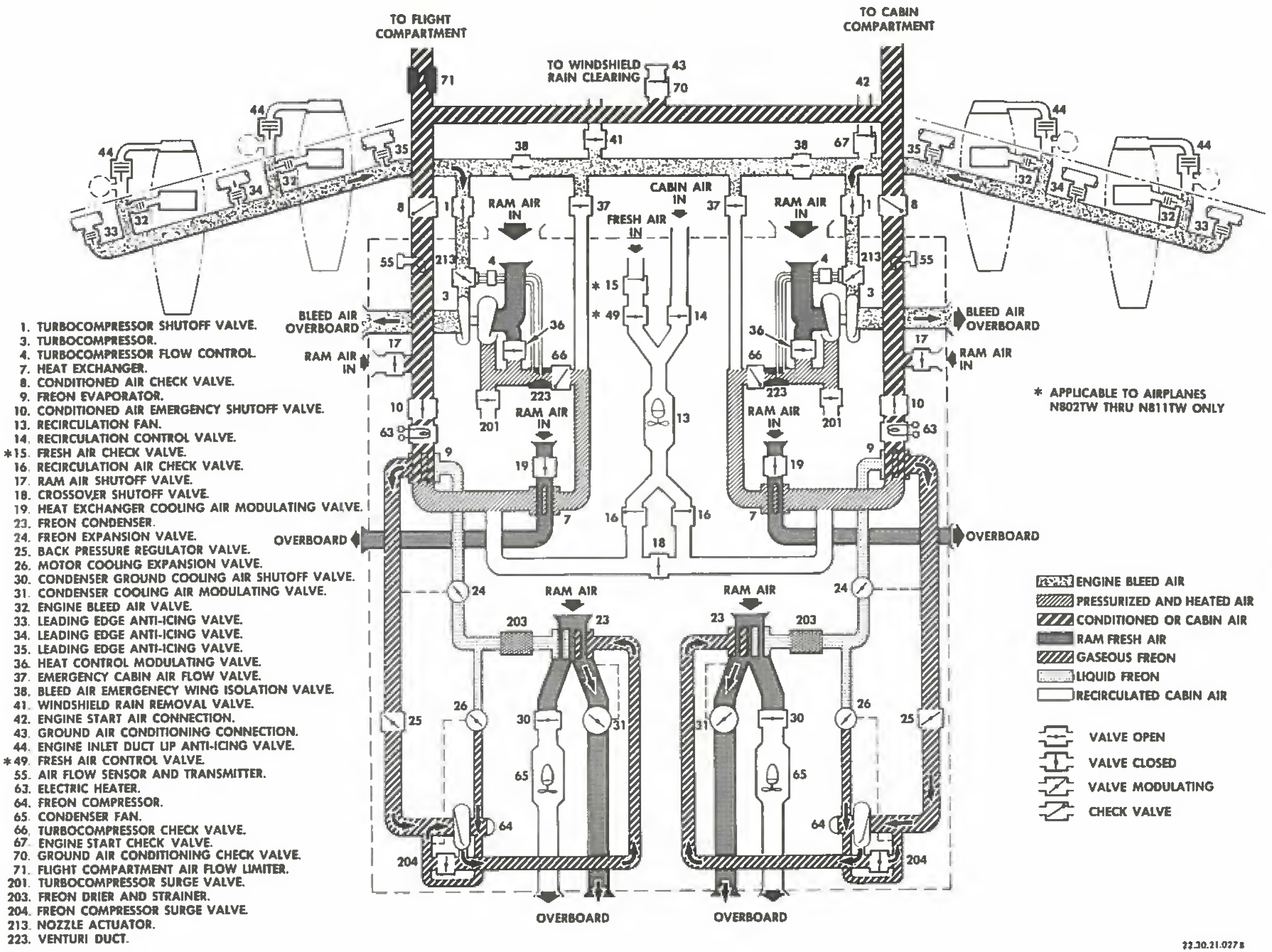
Turbocompressors
Emergency pressurization



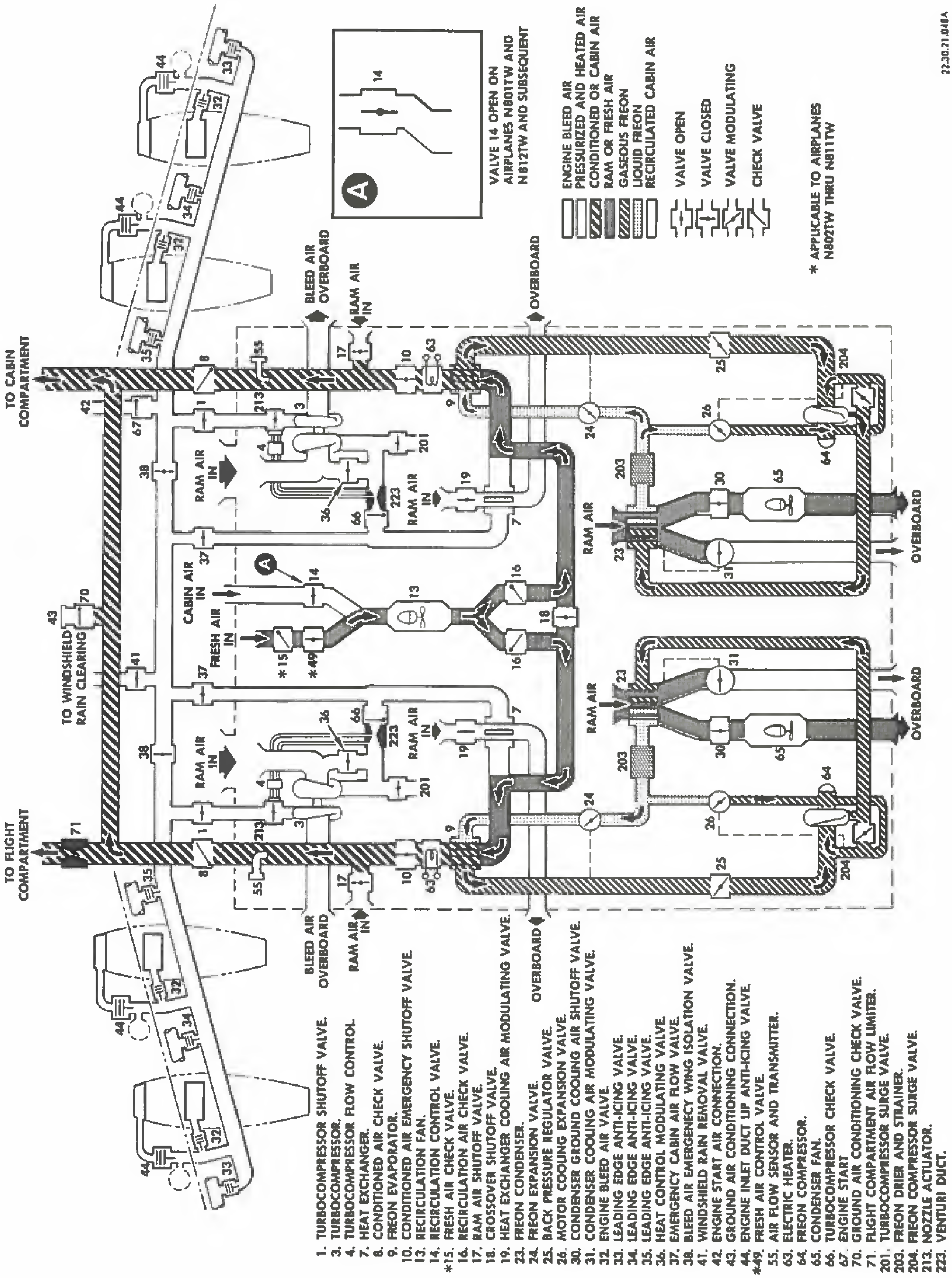
Air Conditioning System Schematic
Inflight Heating
Figure 14-12



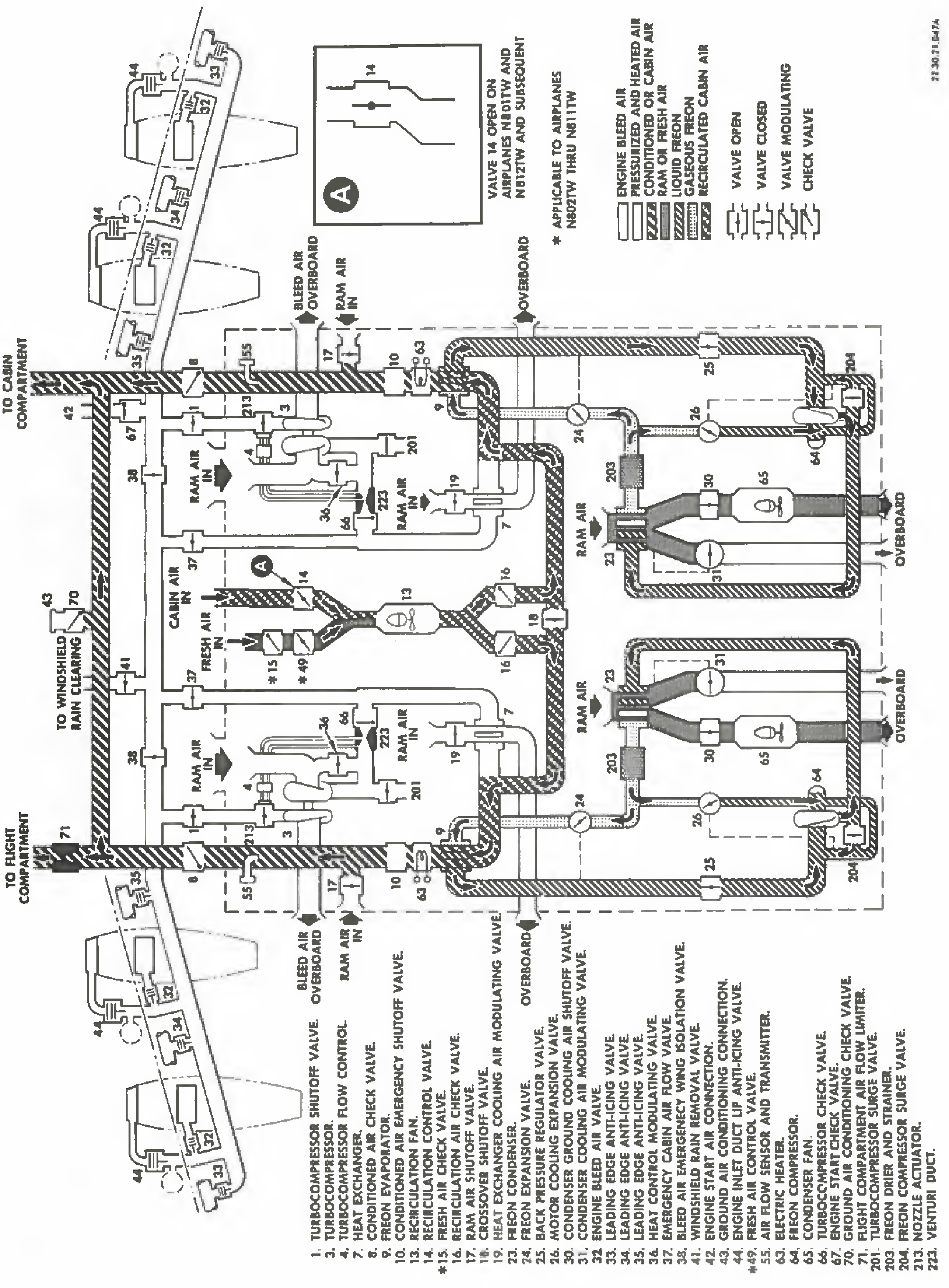
Air Conditioning System Schematic
Inflight Intermediate
Figure 14-13



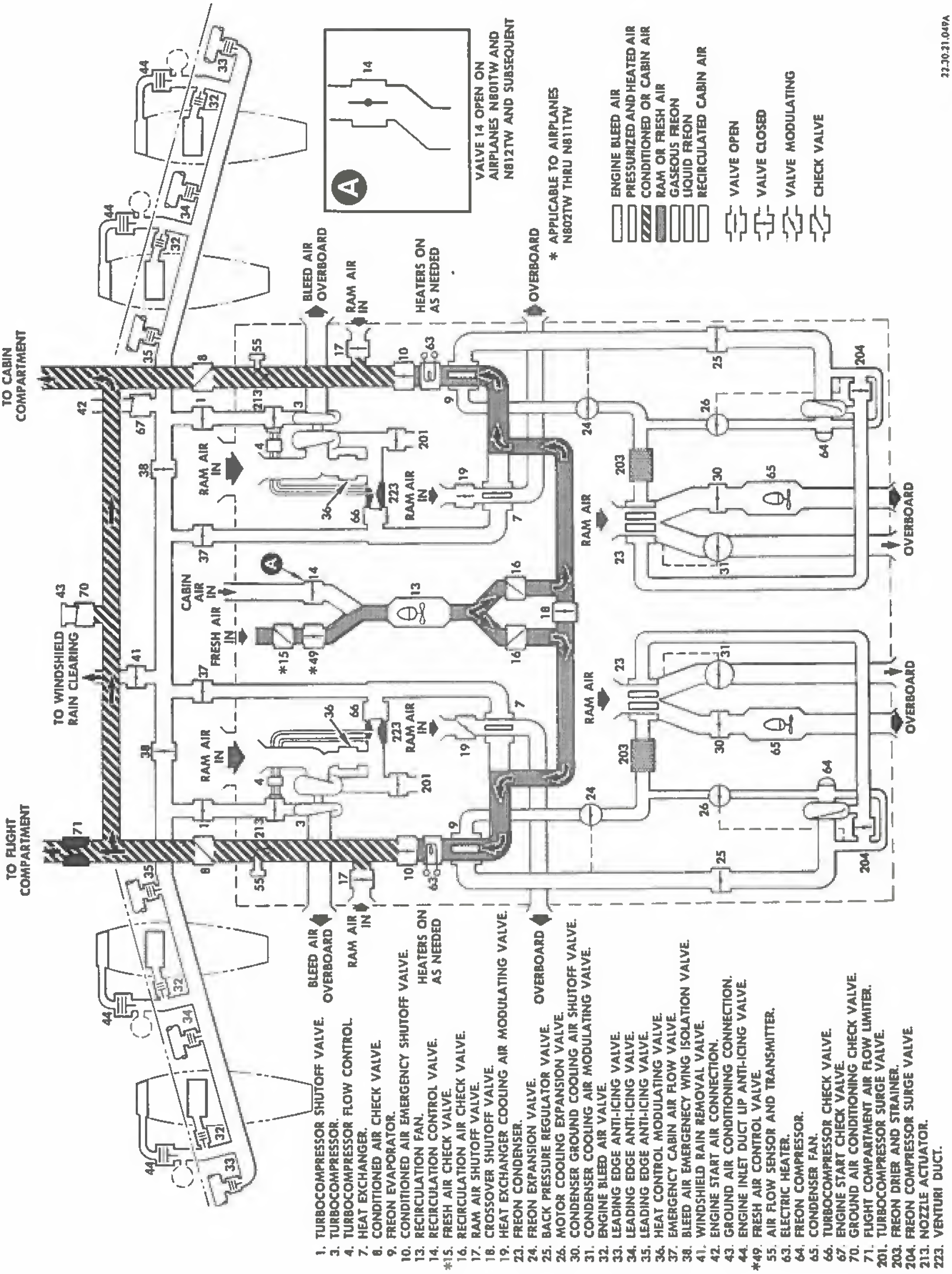
Air Conditioning System Schematic
Inflight Cooling
Figure 14-14



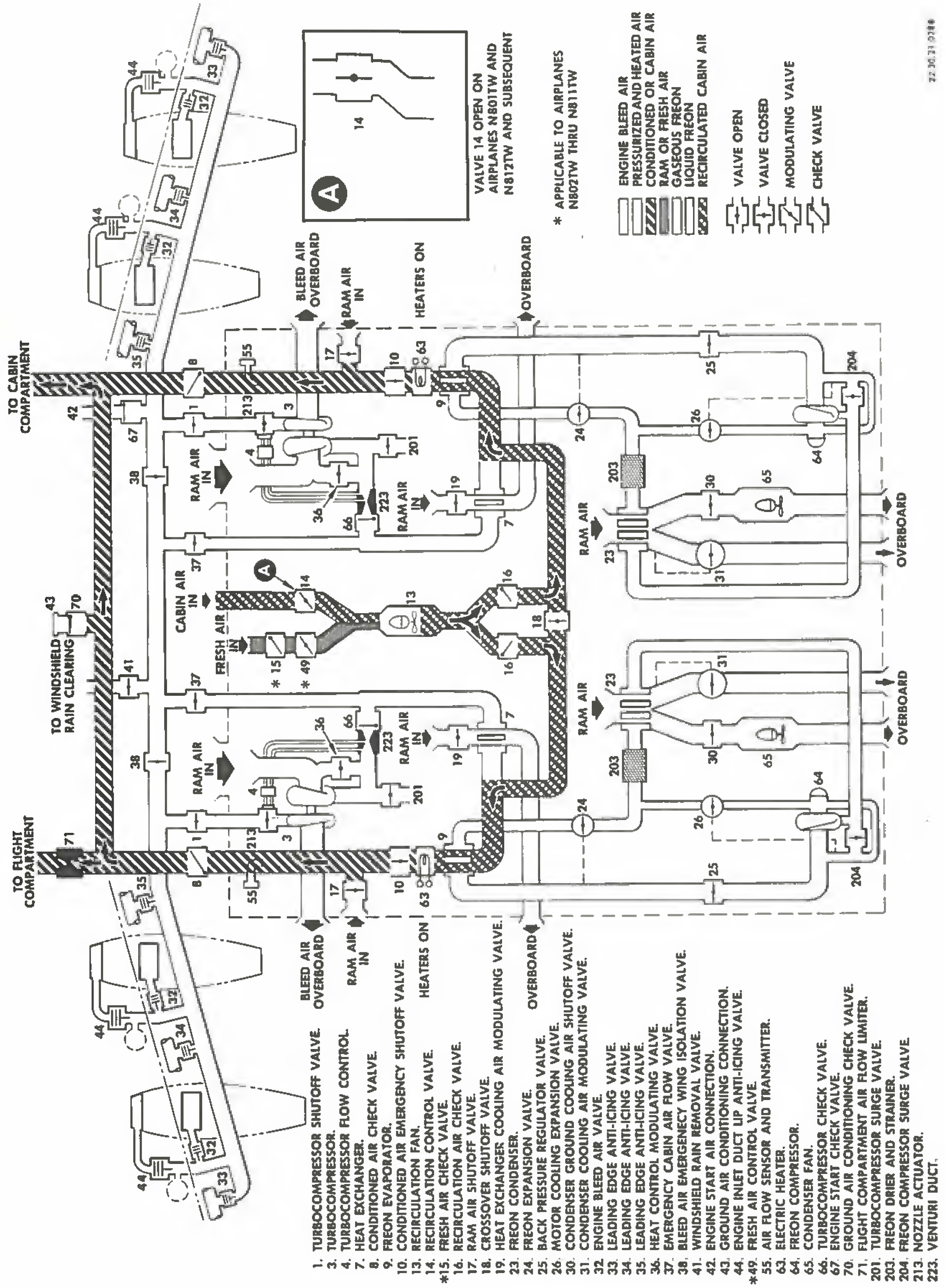
Air Conditioning System Schematic
Ground Cooling
Figure 14-15



Air Conditioning System Schematic
Ground Maximum Cooling
Figure 14-16



Air Conditioning System Schematic
Ground Heating
Figure 14-17



Air Conditioning System Schematic
Ground Maximum Heating
Figure 14-18

AC Operated Equipment

Freon condenser, flight deck
Freon compressor, cabin
Electric heater, cabin system
Electric temperature control
Ram air shutoff valves
Evaporator manifold shutoff valve
Cabin and flight deck air shutoff valves
Freon condenser, cabin
Recirculating blower
Electronic cooling fan
Freon compressor, flight deck
Flight deck ac heaters
Electric heater, flight deck system
Electric temperature control, flight deck

DC Operated Equipment

Warning lights
Temperature and airflow indicators
Turbocompressor shutoff valves and electronic cooling valve
Auxiliary bleed air shutoff valves
Controls for electric heaters, Freon refrigeration units, cabin air recirculating system, electronic cooling fan, and electronic cooling valve
Cabin pressure regulators

Section 15

BLEED AIR AND ADVERSE WEATHER SYSTEMS

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BLEED AIR AND ADVERSE WEATHER SYSTEMS

BLEED AIR SYSTEM

The engine bleed air system provides hot air to operate air conditioning and pressurization equipment, wing anti-icing, engine anti-icing, windshield rain clearing and emergency cabin pressurization. Two overheat protection systems are considered a part of the engine bleed air system, the Leading Edge and Duct Space Temperature system and the Excess Heat and Isolation system.

Bleed Air Sources and Flow Routing

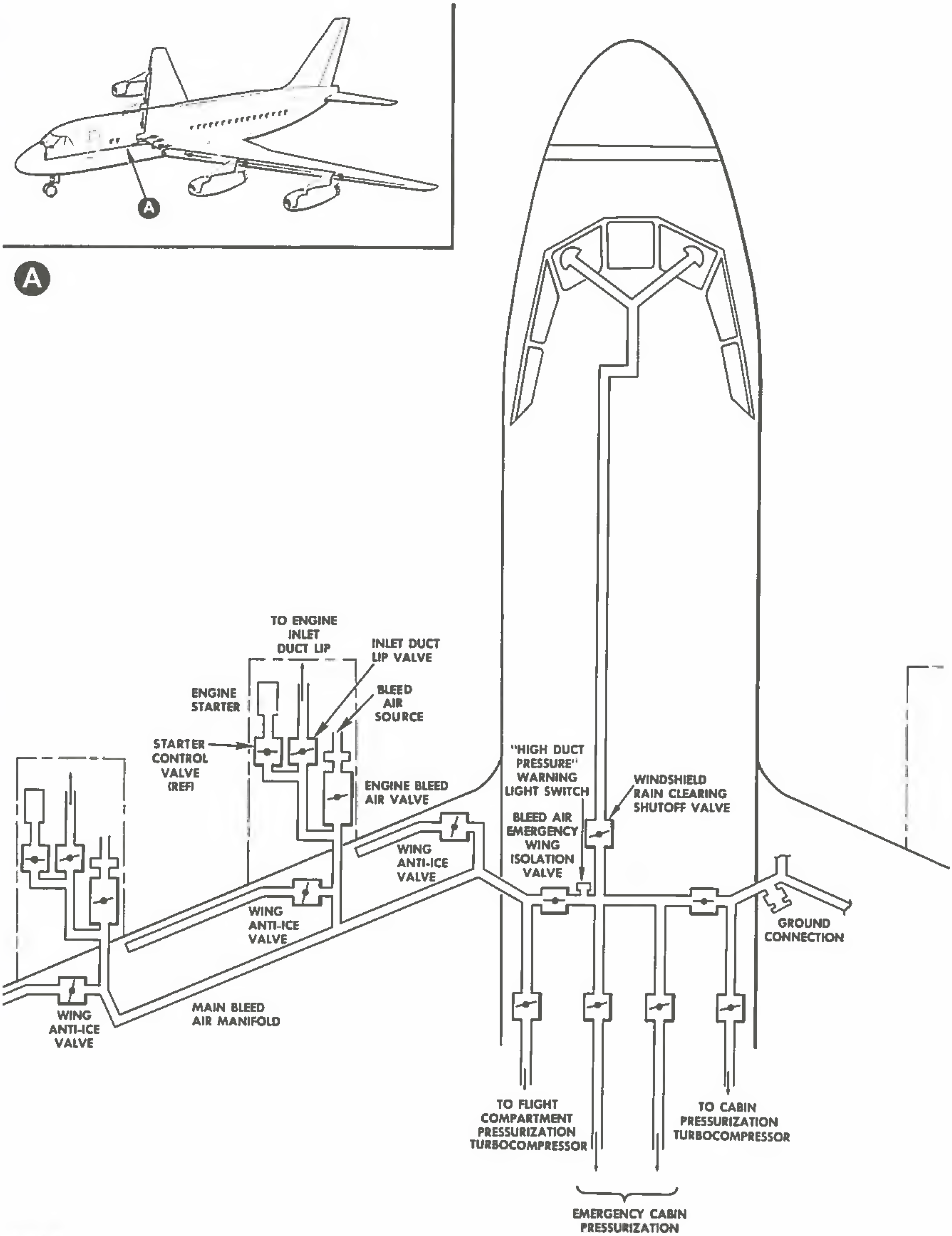
Bleed air is obtained from the seventeenth stage of the engine compressor at 237 psig maximum pressure and 427 degrees C (800 degrees F) maximum temperature. The hot air is manifolded from four ports in each engine into a single duct (see Figure 15-1). A bleed air pressure regulator shutoff valve limits down stream bleed air pressure to 40 psig. It also acts as a check valve to permit engine starting and as a shutoff valve for restricting bleed air at the source. The bleed air is routed through the pod and pylon into the wing bleed air manifold attached to the front spar. The right and left wing bleed air manifolds are connected by a fuselage crossover duct. A bleed air emergency wing isolation valve is located at each end of the fuselage crossover duct. The windshield rain clearing bleed air duct leads forward from the fuselage crossover duct. A windshield rain removal shutoff and isolation valve is provided close to the crossover duct.

Bleed Air Control Switches

Four two-position OPEN-CLOSE toggle switches are installed on the engine bleed air control panel located on the pilots' overhead switch panel. The switches control the operation of the engine bleed air regulator shutoff valves. In the OPEN position, the valves are opened unless closed by overriding signals from the excess heat and isolation system. In the CLOSE position, the valves remain spring-loaded closed regardless of the demands of any airplane system. Engine bleed air switches must be OPEN to use the anti-icing, de-icing, rain clearing and various air conditioning systems (see Figure 15-2).

Bleed Air Control System Indicator Lights

Four amber warning lights, labelled CLOSED, are located above the four engine bleed air control switches. The lights will illuminate when the engine bleed air switches are in the OPEN position and the bleed air valves close due to malfunction or excess heat signals. A single red HIGH DUCT PRESSURE warning light, located above the four amber lights, illuminates if bleed air pressure in the crossover duct increases above 50 psig.



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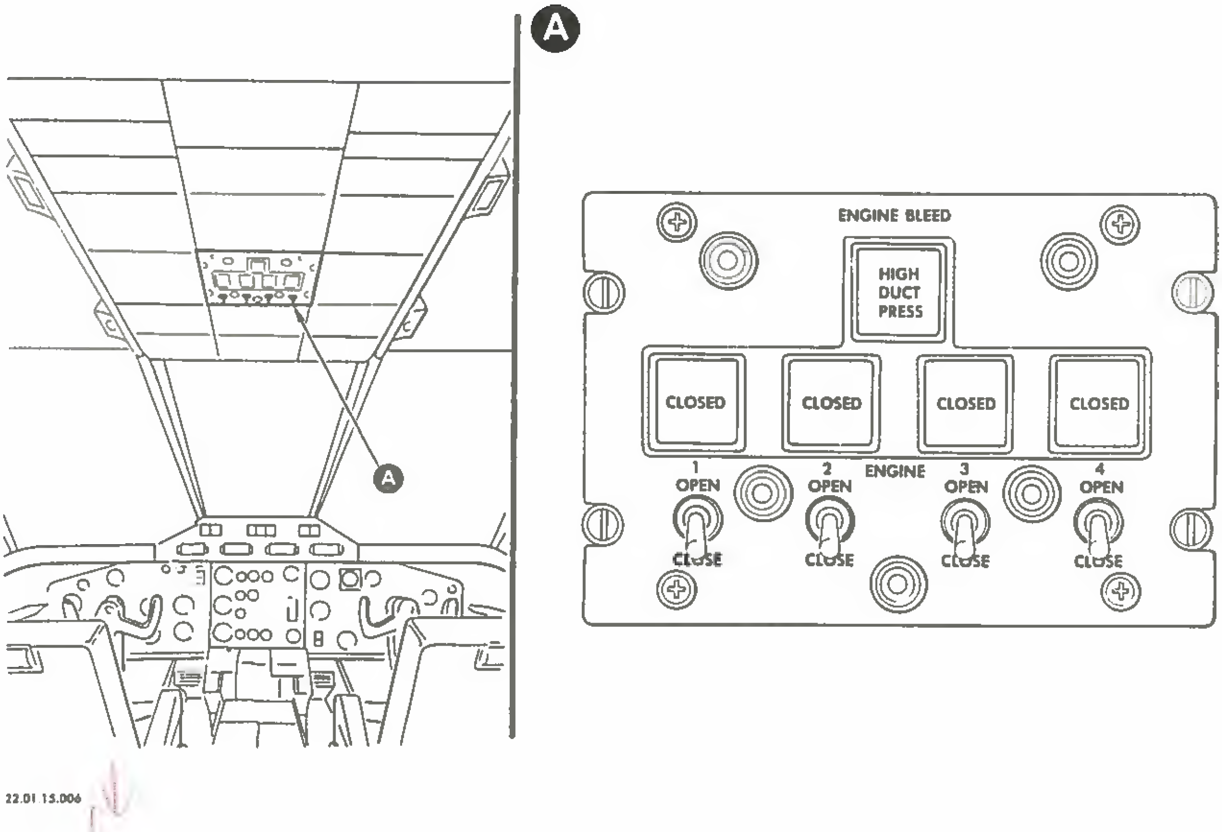
Bleed Air Supply System
Figure 15-1



Engine Bleed Air Control Panel
Figure 15-2

EXCESS HEAT AND ISOLATION SYSTEM

The excess heat and isolation system is a protective system that monitors the ambient temperatures in the wing duct spaces ahead of the front spar and around the rain clearing duct in the fuselage area. Three continuous temperature-sensitive loops are provided. Two follow the basic bleed air duct routing, one loop in each wing. The wing loop paths follow the ducting into the engine pod and pylon areas. The wing temperature-sensitive loops are set to react to a temperature of 154 degrees C (310 degrees F) or higher. The third temperature-sensitive loop follows the fuselage crossover and rain clearing bleed air ducts and is set to react to a temperature of 130 degrees C (266 degrees F) or higher.

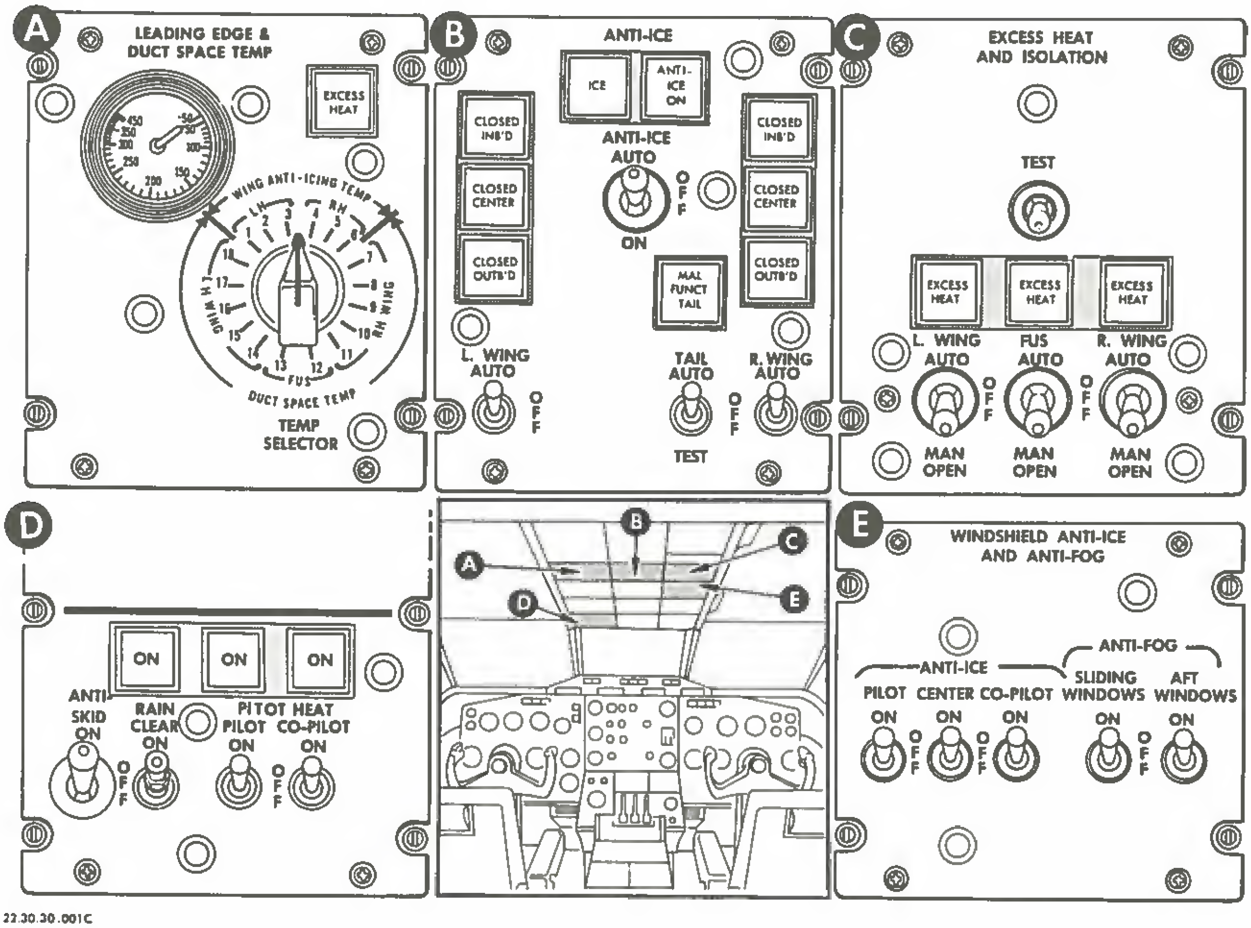
The excess heat and isolation system is designed for inflight operation. The wing temperature-sensitive loops are rendered inoperative while on the ground. The temperatures at which the system will react are set higher than the reaction temperatures of the Leading Edge and Duct Space Temperature system. In normal operation, the latter system will indicate high temperature troubles well in advance of the indications from the excess heat and isolation system. The intent of the design is that the flight crew will be alerted to a high temperature condition, and be able to locate the trouble spot, by using the Leading Edge and Duct Space Temperature system. Should the flight crew miss an over-temperature condition, through other requirements on their time, the excess heat and isolation system will provide warning and automatically isolate the complete area involved in the over-temperature (see Figure 15-3).

Excess Heat and Isolation System Operation

The Excess Heat and Isolation system control panel is located on the pilots' overhead switch panel. Three three-position AUTO-OFF-MAN OPEN switches are located on the panel. Directly above each switch (L WING - FUS - R WING) are three amber warning lights labelled EXCESS HEAT. Each warning light is connected to its respective temperature-sensitive loop. During normal operation, all three switches are placed in the AUTO position. An over-temperature condition in the alarm range of any loop will result in illumination of the associated warning light. If a wing loop is concerned, the engine bleed air valves for both engines in that wing and the emergency isolation valve will close, thus shutting off bleed airflow through the affected wing.

NOTE: As a result of this valve action, the engine bleed air warning lights for the engines concerned will illuminate. If the airplane anti-icing systems are operating, the engine anti-ice warning lights for the engines concerned and the three anti-ice wing valve warning lights for the wing concerned will also illuminate.

If an over-temperature condition in the fuselage crossover or rain clearing duct is involved, the EXCESS HEAT light will illuminate. At the same time, both wing emergency isolation valves will close. The engine bleed air valves will not close. The turbo compressors, being supplied from outboard of the isolation valves, will remain in operation.



Ice Protection System Controls
Figure 15-3

NOTE: As a result of this action, if the airplane adverse weather systems are operating, the windshield rain clearing duct shutoff valve will close from lack of bleed air pressure, thus extinguishing the rain clearing ON light.

Should the external icing conditions be such that manual override of the automatic heat isolation system is desirable, the appropriate Excess Heat and Isolation system control switch can be placed in the MAN OPEN position for short periods of time. Placing any of the switches in MAN OPEN will open all isolation and engine bleed air regulator valves that have been closed automatically. Placing any of the switches in the OFF position will extinguish the appropriate warning lights and position the isolation and bleed air valves concerned to the closed position. Thus in the OFF position, the same valves are closed as are closed in the AUTOMATIC position by an overheat condition. However, the EXCESS HEAT warning light is extinguished.

CAUTION: PLACING AN ISOLATION SWITCH IN THE MAN-OPEN POSITION BYPASSES THE AUTOMATIC OVERHEAT PROTECTION AND EXCESS HEATING OF THE AIRPLANE STRUCTURE MAY RESULT. PLACING THE FUSELAGE SWITCH IN MAN-OPEN WILL OVERRIDE BOTH AUTOMATIC AND MANUAL CLOSING OF THE WING ISOLATION VALVES. THIS WILL MAKE IT IMPOSSIBLE TO ISOLATE ONE WING SINCE CROSSFEED OCCURS FROM THE OTHER WING.

LEADING EDGE AND DUCT SPACE TEMPERATURE SYSTEM

The Leading Edge and Duct Space Temperature system provides a means of monitoring temperatures at 18 places throughout the wing. Twelve of the temperature sensing units are thermistor types. Five are located in each wing leading edge, spaced along the front spar in the bleed air duct area. Two are located in the fuselage section forward of the front spar and along the rain clearing bleed air duct. The wing area thermistors are set at an alarm temperature of 149 degrees C(300 degrees F) and the two forward fuselage units are set at a temperature of 65.6 degrees C (150 degrees F). Six skin patch type temperature sensors are located, one to each section, in each wing leading edge. These units are inserted in a cutout in the leading edge at the point where bleed air temperature will be at its highest when entering the distribution tubes. Each patch consists of two separate sensor units. One automatically controls its associated wing anti-ice valve and the other is connected to the overheat alarm system.

Leading Edge and Duct Space Temperature System Operation

The Leading Edge and Duct Space Temperature system control panel is located in the pilots' overhead switch panel. The rotary eighteen-position switch connects

265° (80
130°C

each temperature sensing unit to a temperature indicator calibrated in degrees F. The switch sectors are divided into two major divisions, WING ANTI-ICING TEMP and DUCT SPACE TEMP. The WING ANTI-ICING TEMP is divided into LH and RH segments, each containing three temperature points. The DUCT SPACE TEMP segment divides into five LH WING, two FUS and five RH WING temperature points. An amber EXCESS HEAT warning light is provided and the light will illuminate when any temperature-sensitive element detects a temperature condition in excess of its alarm setting. The particular element sending the signal can be located by checking the temperature of the eighteen points. Manual isolation of the overtemperature section may then be accomplished.

WING SPAR VENTILATION SYSTEM

The face of each wing front spar and a fiberglass baffle attached to the wing leading edge enclose an area in which the wing bleed air manifold is routed. The areas around the manifolds are vented by ram air entering through two flush type inlets located on the lower surface of the wing adjacent to the inboard engine pylons. Half of the air flows outboard and is discharged through ports on the lower surface of each wing near the tip, while the remainder of the air flows inboard and is ducted into the fuselage and dumped overboard through a discharge port (see Figure 15-4).

Ventilating air removes any fuel vapors which may be present, and limits the maximum temperature on the outside of the bleed air ducting to approximately 135 degrees C (275 degrees F).

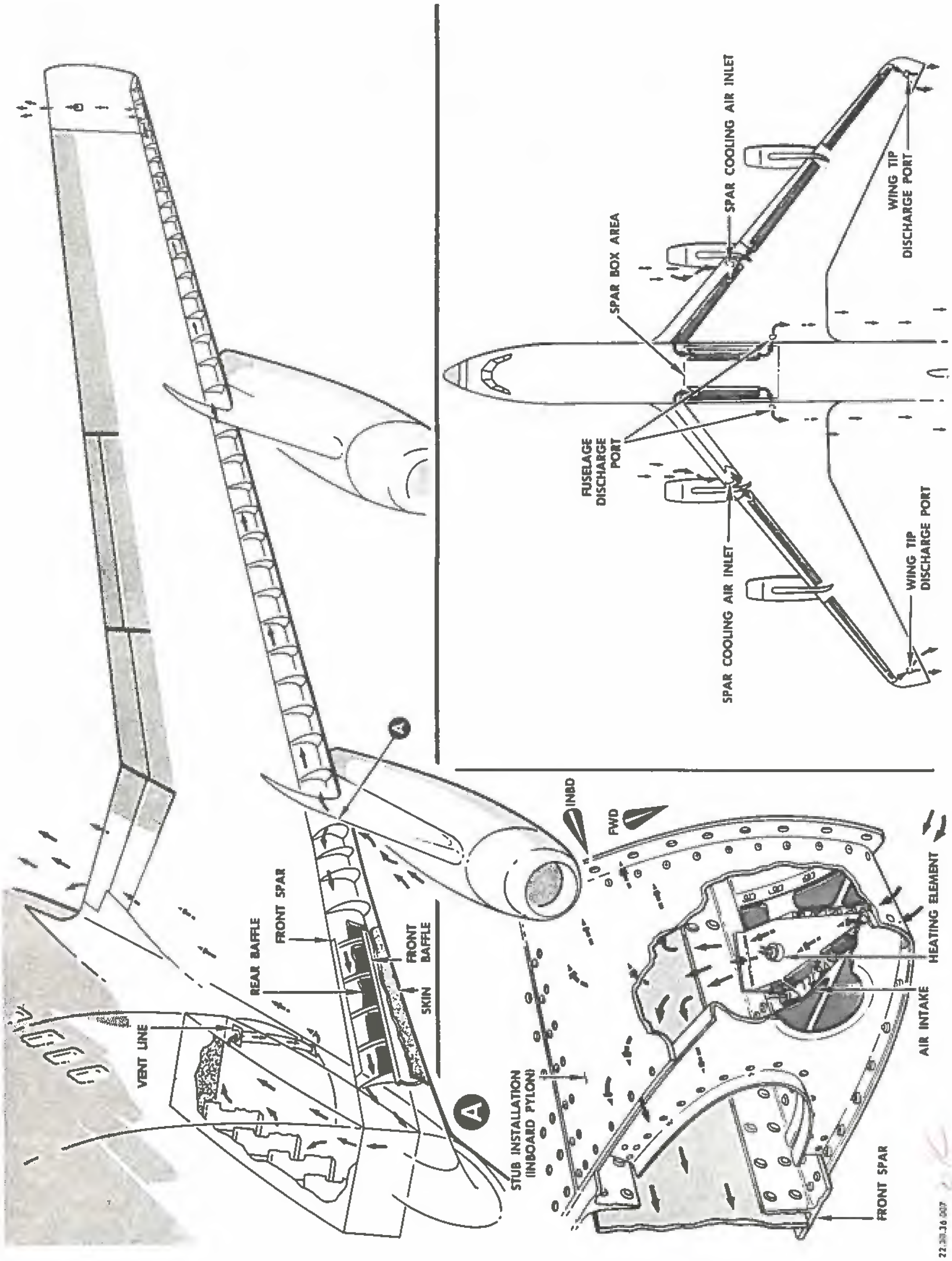
Ventilation Duct Lip Anti-Icing

The ventilation inlet duct lip is protected from ice by electric heaters. These heaters are controlled by the right and left wing switches on the anti-ice control panel and respond to signals from the ice detector probes when the switches are in the AUTO position. They remain on continuously as long as an icing signal is being received from the ice detectors.

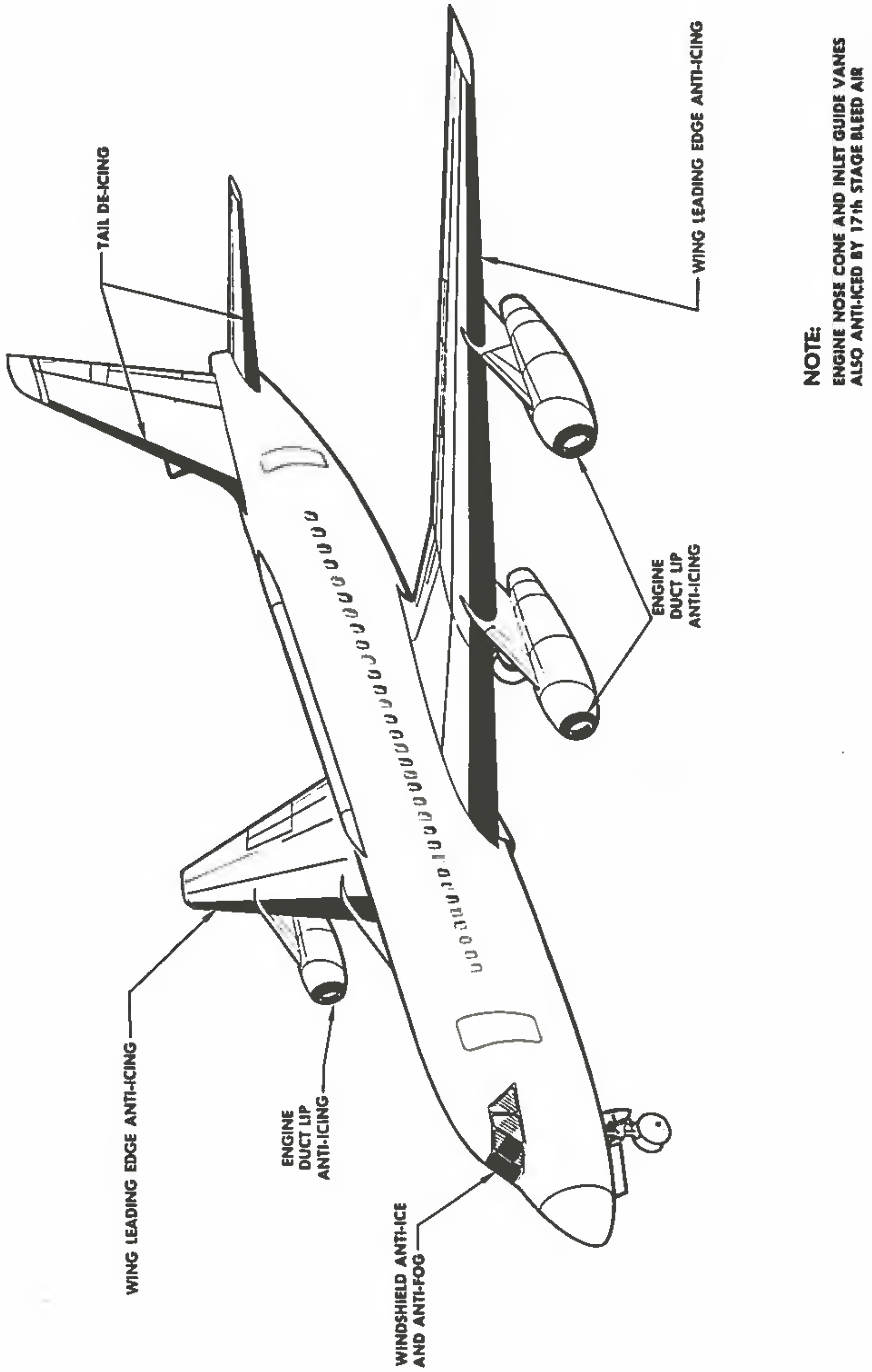
ADVERSE WEATHER SYSTEMS

Anti-icing and de-icing systems, an automatic ice detection system, and a windshield rain removal system provide complete adverse weather protection. The ice and rain protection systems are as follows: (Figure 15-5)

1. Thermal anti-icing system for the wing leading edges.
2. Thermal de-icing system for the tail leading edges.
3. Thermal anti-icing and anti-fogging systems for the windshield.
4. Pneumatic rain clearing system for the windshield
5. Thermal anti-icing system for the pitot system probes.
6. Thermal anti-icing system for engine air inlet areas.



Wing Spar Ventilating System
Figure 15-4



Adverse Weather Systems
Figure 15-5

The wings are protected against ice formation by a double-skin, hot-air, anti-icing system installed in the wing leading edges. The tail de-icing system uses integral, electrically heated blankets on the horizontal and vertical stabilizer leading edges. The center and main windshields are anti-iced by electrically heating a high density transparent conductive coating on the inner surface of the outer glass ply. The main windshields are anti-fogged by electrically heating a low density coating parallel to the high density coating. Anti-fogging of the sliding and aft windows in the flight compartment is accomplished by electrically heating a low density conductive coating on the outer surface of the inner glass ply.

The rain clearing system routes hot air to the windshield area and injects it parallel to the airstream through nozzles. The pilot system probes on each side of the fuselage are thermally anti-iced by electrical heating units integral with the probe heads. The engine air inlet duct lip, engine compressor nose cone and guide vanes on each engine are anti-iced by the application of hot air.

Control of the anti-icing and de-icing systems is normally automatic, however, manual control by the pilots is provided for in all systems. All control switches for the anti-icing and de-icing systems are located on the pilot's overhead switch panel (see Figure 15-2).

AUTOMATIC ICE DETECTION SYSTEM

The basic control element of the anti-icing and de-icing systems is the automatic ice detection unit. When the airplane anti-icing and de-icing systems are operated in their automatic mode, the ice detector will sense an icing condition, turn on all systems, and turn the systems off when they are no longer needed. The heart of the ice detector system is the ice detector probe.

Ice Detector Probes

An ice detector probe is located in the engine air inlet duct of engine No. 1 and No. 3. This location was selected since the engine air inlet guide vanes are a critical component from an icing standpoint. When icing conditions are encountered, pressure openings in the detector probes fill with ice and close. The resultant change in air pressure within the probe operates a pressure differential switch which in turn actuates relays to energize the ice protection systems and begin the de-icing cycle of the ice-detector probes.

Detector Probe Operation

The ice detector probe is composed of two elements. The reference probe, connected to one side of the pressure diaphragm, is maintained at a temperature above the freezing point of water at all times. A 175-watt heater turns on at

temperatures below 43 degrees C (110 degrees F) and off at temperatures above 49 degrees C (120 degrees F). When ice forms on the cold detector probe, connected to the other side of the diaphragm, it covers the pressure sensing orifices in the probe, thus creating a pressure differential. The resultant pressure differential actuates a switch which turns on all ice protection systems whose switches are in automatic position, and also energizes a time delay circuit and heater element in the cold probe.

The heater element sheds the accumulated ice in approximately eight seconds. The time delay unit keeps the ice protection systems in operation for one minute, during which time the ice sensing probe heater element is off and the probe can again accumulate ice. If another ice signal is transmitted to the control unit during the latter part of the one-minute period, the timer resets itself to zero and the on signal is continued to the ice protection system. At any time during the cycling periods that an ice signal is not received, the control circuit shuts off the ice protection systems (see Figure 15-6).

ANTI-ICE MASTER SWITCH

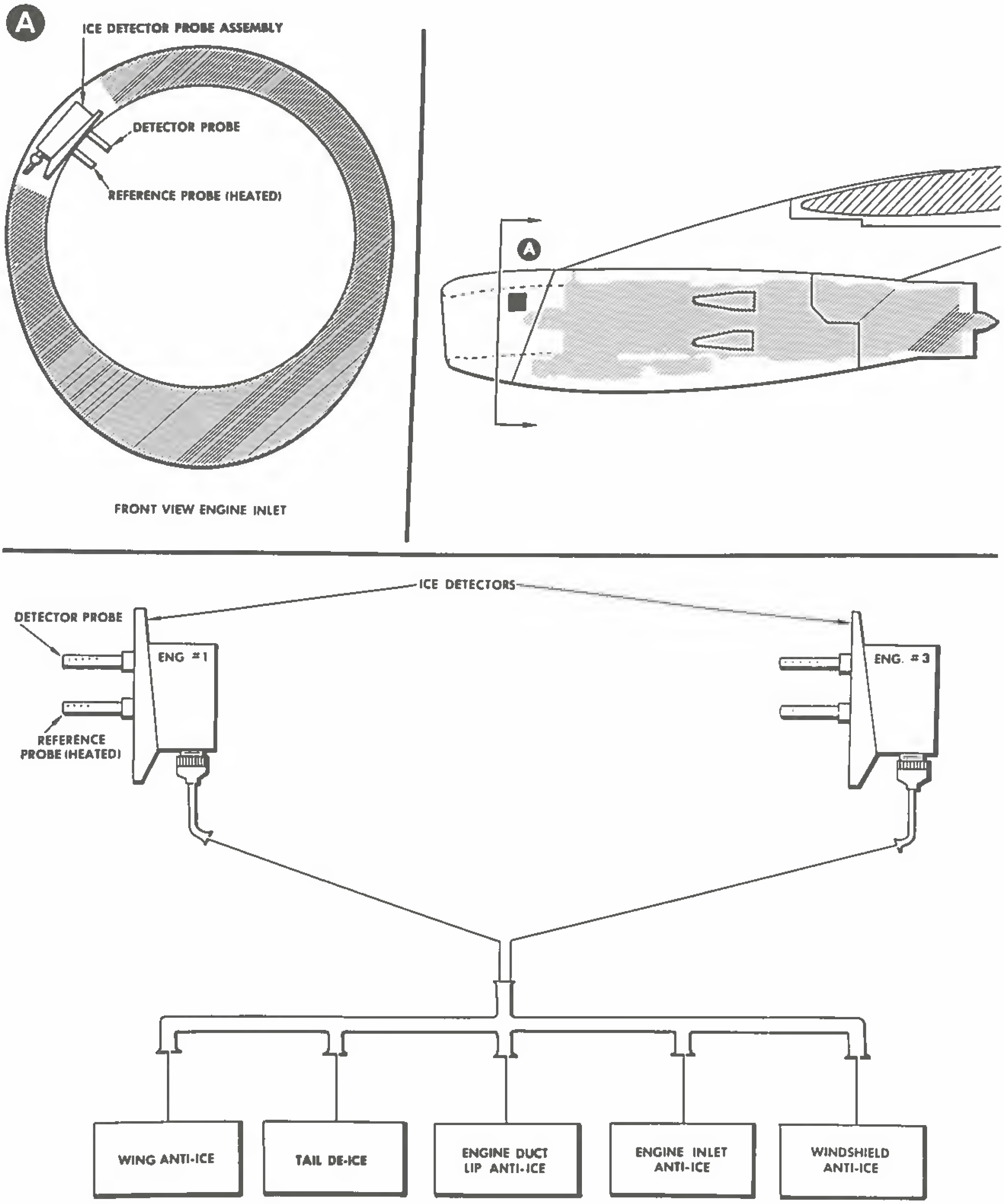
A three position AUTO-OFF-ON master anti-ice toggle switch is located in the center of the anti-ice control panel. In normal operation, the wing anti-ice, engine anti-ice, and tail de-ice switches are all placed in their AUTO positions. This action connects all the systems into the master switch. Placing the anti-ice master switch in AUTO position arms the automatic ice detection system and arms all connected systems ready for operation as soon as an ice detection signal is received. When the master switch is in the ON position, all connected anti-ice systems are turned on, including windshield anti-ice, regardless of their individual switches being in the AUTO position. In the OFF position, the master switch de-energizes the ice-protection systems, regardless of the individual switch positions. The switch must be pulled out to clear a detent before it can be moved from any selected position. (The engine anti-ice switches bypass the master ice switch.)

Anti-Ice Master Switch Indicator Lights

Two indicator lights are provided above the master switch. A red ICE light and a blue ANTI-ICE ON light. When the master switch is in the AUTO position, and the ice detector senses ice, the red ICE light will illuminate if any of the anti-icing switches (left wing, right wing, tail and engine) are not in the AUTO position. The blue ANTI-ICE ON light illuminates when a signal is received from the ice detectors and all anti-ice system switches are in AUTO position.

WING ANTI-ICE SYSTEM

Each wing leading edge is divided into three sections for anti-icing purposes: an inboard section between the inboard pylon and the fuselage, a center section between the inboard and outboard pylons, and the outboard section between the outboard pylon and the wing tip. Each of the six sections receives bleed air through the six wing anti-ice valves located in ducts connected to the bleed



Ice Detector Probes (Typical)
Figure 15-6

air system. The wing anti-ice valve reduces the 40 psig duct pressure to 13 psig and passes it into the distribution tube where it is ported to the leading edge skin passages. The distribution tube ports are small drilled holes which reduce the 13 psig bleed air pressure to approximately 2 psig. After passing chordwise through the leading edge passages, the air is dumped into the area between the forward and aft baffles in the leading edge just forward to the spar. The air is then dumped overboard through slots located on the lower surface of the leading edge at the front spar (see Figure 15-7).

Wing Anti-Ice Valves

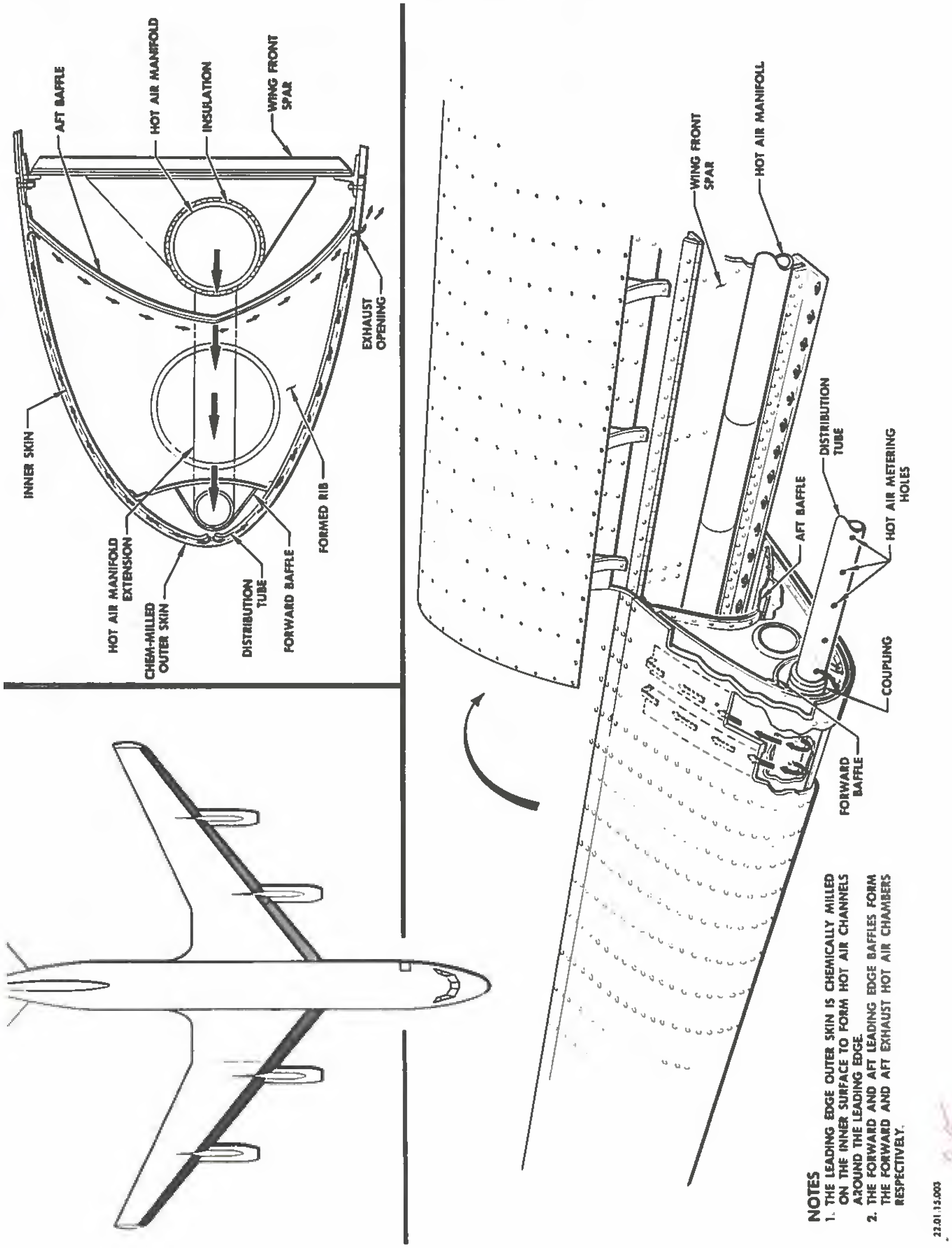
In addition to regulating the flow of bleed air to the distribution tubes, the six wing anti-ice valves can be closed automatically or manually when an overheat condition exists. The six skin patch temperature sensors (those paired with the six patches connected to the leading edge and duct space temperature system) are connected to their associated wing anti-icing valve. The valve control sensor will, upon detecting a skin temperature of 157 degrees C (315 degrees F), cause its associated wing anti-icing valve to close. Upon sensing a temperature drop to 46 degrees C (115 degrees F), the valve control sensor will re-signal the wing anti-ice valve to open. No provisions have been made for manual opening of any wing anti-icing valves against a closing signal from the sensor patches. However, all the valves (3) in each wing can be manually closed by placing the appropriate wing anti-icing switch in the OFF position.

Wing Anti-Ice Switches

Two two-position AUTO-OFF switches are provided for wing anti-icing control. One is located in the lower left side of the anti-ice panel, labeled L. WING, and one in the lower right side labeled R. WING. In AUTO position, with the master switch in the AUTO position, the wing anti-ice shutoff valves are controlled by signals from the ice detector. With the wing anti-ice valve switches in AUTO position, and the master switch in ON position, the wing anti-ice valves are opened. When the anti-ice valve switches are in OFF position, the valves are spring-loaded closed. A WING ICE LTS switch, in the EXTERIOR LIGHTS control panel, activates wing leading edge flood lights for night inspection purposes.

Wing Anti-Ice Indicator Lights

Directly above each wing anti-ice switch are three amber lights labeled CLOSED INBD, CLOSED CENTER and CLOSED OUTBD. Each light represents its respective wing anti-ice valve. With the wing anti-ice switches in AUTO position, the master switch in AUTO position, and the ice sensing signal applied, a light will illuminate if a valve fails to open or is closed by an overheat condition. The lights operate in a similar manner with the master switch in an ON position.



Wing Leading Edge
Anti-Ice Air Flow
Figure 15-7

CAUTION: OPERATING THE WING ANTI-ICE SYSTEM IN A MANUAL MODE WHILE ON THE GROUND IS NOT RECOMMENDED DUE TO THE POSSIBILITY OF OVERHEATING THE LEADING EDGES, FRONT SPARS OR VENT SCOOPS. OPERATION FOR TEST PURPOSES UP TO 20 SECONDS IS PERMISSIBLE.

NOTE: Automatic operation is possible only in flight. A pressure safety switch is connected through a landing gear circuit to prevent automatic operation on the ground.

TAIL DE-ICING SYSTEM

The tail de-icing system employs integral cyclic-electrical leading edge heating blankets for periodic removal of ice (see Figure 15-8). The cyclic heating blankets are separated by parting strips which are heated continuously when the system is energized. Eighteen heaters are used, each having an area of approximately 400 square inches. A controller cycles power to the individual heaters for a maximum heat-on time of 12 seconds per segment, and a fixed heat-off time of 180 seconds. This sequence provides optimum de-icing performance with minimum runback ice formation behind the heated area.

Tail De-Icing Overheat Protection

Overheat protection is provided by a microswitch on the main landing gear which prevents operation of the system on the ground, and by a thermoswitch which turns the system off at ambient air temperatures above 7 degrees C (45 degrees F).

Heat Sensors

There are 36 temperature sensors in the heater system, two embedded in each heater pad. Only one sensor in each heater pad is used in the circuit; the others are spares that may be used to replace failed units without the removal of the boot installation. The sensors signal area temperature to the controller. They are set at a value of 10 degrees C (50 degrees F) and, upon reaching this temperature, will shut down the respective heater pad and switch power to the next heater.

Tail De-Icing Controller Unit

An electro-mechanical controller is located in the tail area aft of the pressure bulkhead. Upon receipt of a 28-volt dc signal from the ice detector, the controller closes a 3-phase, 115-volt ac circuit, energizing the parting stips and connecting the No. 1. heating pad to its appropriate ac terminal. A full voltage is then applied to the No. 1 pad for a maximum of 12 seconds. The voltage then returns to zero, the No. 1 heating pad is removed from the circuit, and the No. 2 heating pad is connected to its appropriate terminal. This process is repeated until all 18 heater pads have been cycled.

CYCLED AREAS

FIN

SECTION NO.	POWER INPUT
3	18
6	18
9	18
12	18
15	18
18	18

LH STABILIZER

SECTION NO.	POWER INPUT
1	18
4	18
7	18
10	18
13	18
16	18

RH STABILIZER

SECTION NO.	POWER INPUT
2	18
● 5	18
8	18
11	18
14	18
17	18

PARTING STRIPS (CONTINUOUS)

FIN

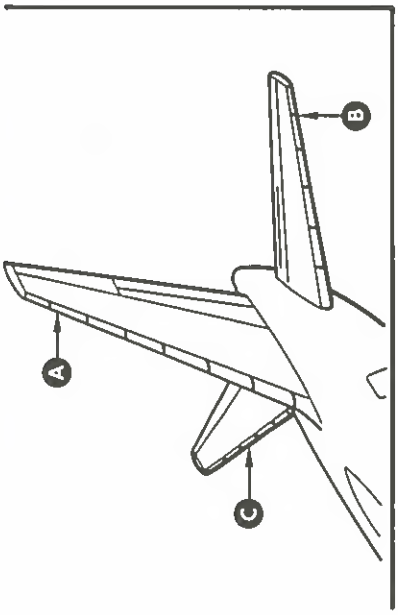
SECTION NO.	POWER INPUT
19	12
20	12
21	12
22	12
23	12
24	12
25	12
26	12
27	12

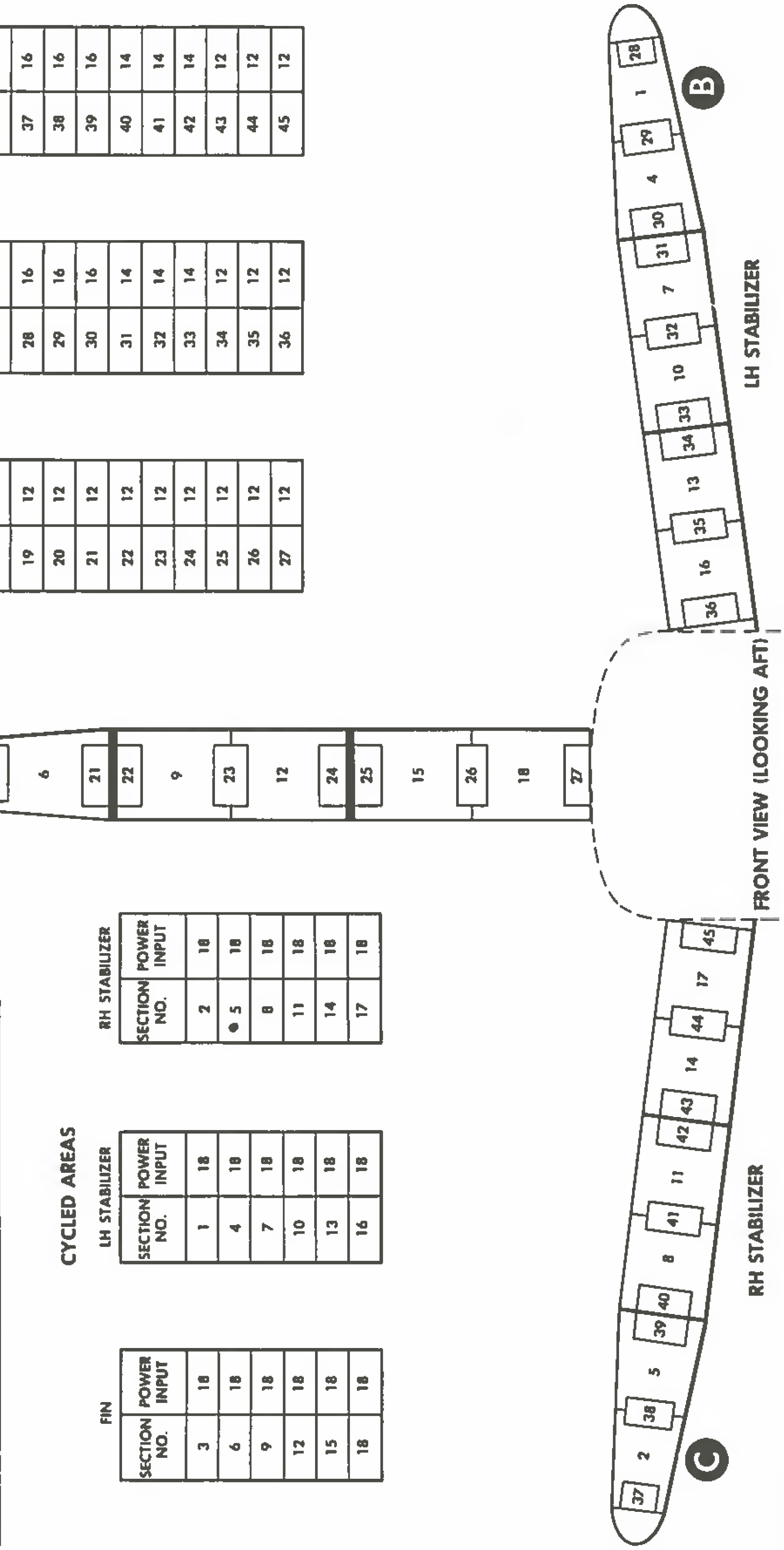
LH STABILIZER

SECTION NO.	POWER INPUT
28	16
29	16
30	16
31	14
32	14
33	14
34	12
35	12
36	12

RH STABILIZER

SECTION NO.	POWER INPUT
37	16
38	16
39	16
40	14
41	14
42	14
43	12
44	12
45	12





Tail De-Icing Heater Pads
Figure 15-8

The time interval of voltage application to the heater pads will vary from a maximum of 12 seconds to some shorter interval, depending on the ambient temperature. The controller will also select individual sensor signals which are synchronized to correspond with the heating pad selected.

Normal time for a complete cycle is 180 seconds. If, however, due to the ambient temperature the individual heater pad operation times are shortened, the controller will shut off cycling until 180 seconds have elapsed from the start of the heating cycle. However, ac power will continue to be impressed upon the parting strips throughout the entire 180 second period.

If the controller fails to receive an impulse from the ice detector upon the completion of the heating cycle or after 180 seconds have elapsed from the start of the heating period, current will be removed automatically from the parting strips and heating pad cycling will discontinue. The controller will then remain in the armed or ready position until it receives another impulse from the ice detector. The unit will cease to operate if ac power fails during a heating cycle. The unit will resume normal operation upon restoration of ac power.

Controller Overload Protection

The controller incorporates overload protection of all three phases of both the parting strips and the heater pads and will disconnect any overloaded phase or phases from the power supply. The parting strip heater circuit protection device provides visual evidence of the overloaded phase. It can be reset by manual means only at the controller itself. The heating pad circuit can be automatically reset by the controller when it selects the first subsequent heating pad that caused the overloaded condition. Fuses are not used in the overload protection devices. Any malfunction of the controller will not result in sustained heating of a heater pad which might cause damage.

Tail De-Icing Control Switch

The tail de-icing system is controlled by a toggle switch located in the anti-ice section of the pilot's overhead switch panel. This switch is labeled TAIL and has three positions, AUTO-OFF-TEST. In the AUTO position, the tail de-icing system is controlled by the automatic ice detector system. An amber light labeled MALFUNCT-TAIL is located directly above the switch.

Tail De-Icing Malfunction Indicator

The MALFUNCT-TAIL warning light will illuminate during the following malfunctions:

1. Overload of any phase in the parting strip circuits by continuous indication while the controller is operating.
2. Overload of any phase of the heater pad circuits by an indication only while the faulty pad is selected for heating.
3. Failure of the controller motor to start.
4. During ground test of the de-icing system.

The controller circuit may be tested prior to flight by placing the control switch in the spring-loaded momentary contact TEST position and observing 18 "blinks" on the MALFUNCT-TAIL light over a 10 to 15 second period of time.

NOTE: The TEST mode cannot be used in flight. It is usable only for ground testing purposes.

WINDSHIELD ANTI-ICE AND ANTI-FOG SYSTEMS

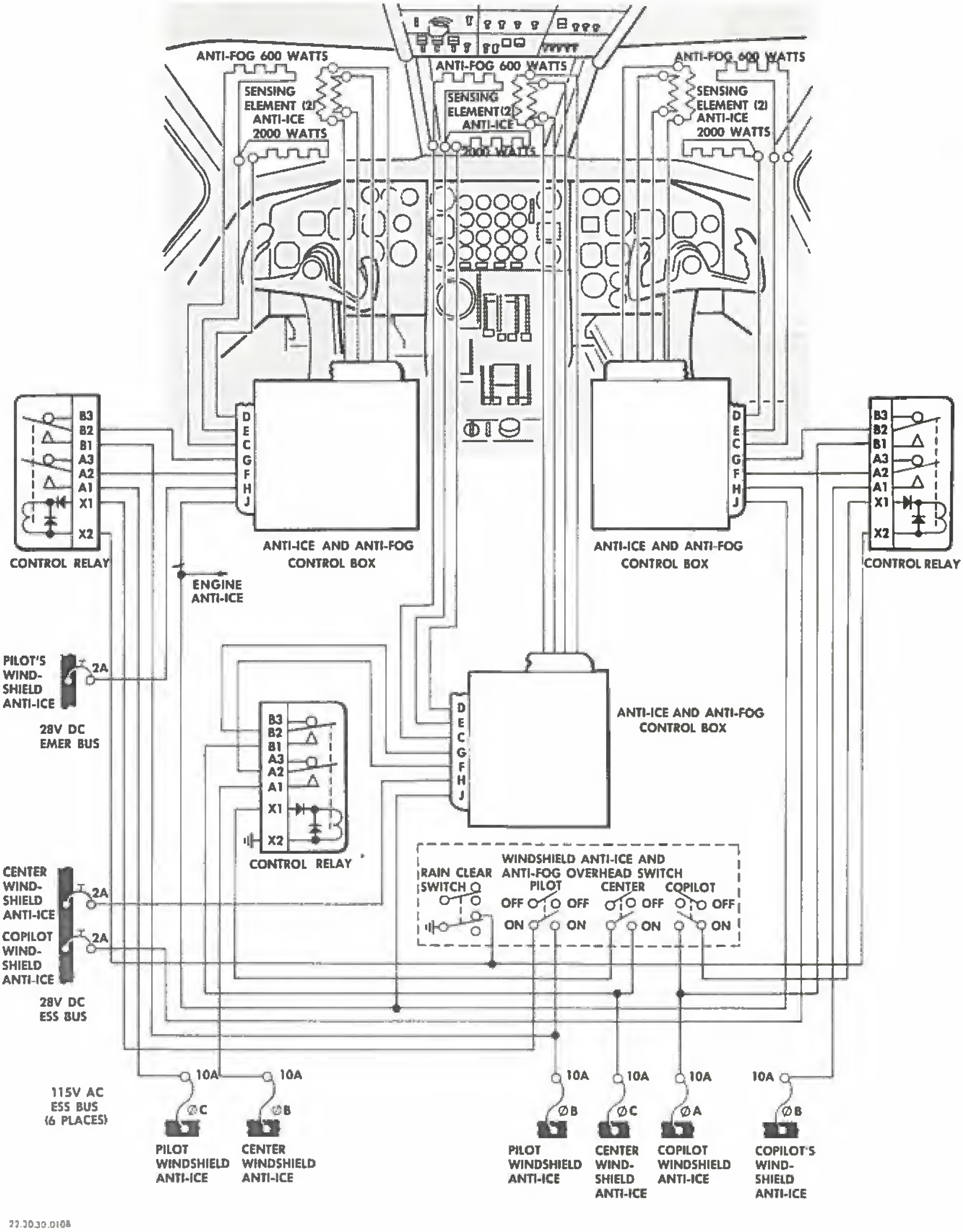
The windshield anti-icing and anti-fogging systems provide optimum visibility under extreme weather conditions. Five-ply sandwich construction consisting of three layers of glass separated by two layers of vinyl plastic is used in both the windshields and side panels. Windshield anti-icing is accomplished by supplying electrical current to a high-density conductive coating on the inner surface of the outer glass panels. Anti-fogging for the windshields is provided by a parallel low-density coating on the outer surface of the inner glass panels. The side panels and sliding windows are provided with anti-fogging only, accomplished by the same method as that used for the windshields (see Figure 15-9). The anti-fogging system is designed to operate at all times during flight. The heat supplied to the low-density coating of the windshields maintains the aft vinyl layers at approximately 38 degrees C (100 degrees F) and tends to increase the structural integrity of the windshields, making them more resistant to impact.

Windshield Overheat Protection

When icing conditions are encountered, the ice detector system automatically energizes the anti-ice heating circuits and de-energizes the anti-fog circuits on the windshield panels. Both the windshield anti-icing and anti-fogging coatings are protected against overheating by sensing elements located in the vinyl layer next to the conductive coating. The sensing element controls the heat input by maintaining a constant temperature, thus precluding an overheated windshield. Protection against overheat by the rain clearing system is also provided.

Windshield Anti-Ice-Fog Control Switches

The windshield anti-icing and anti-fogging control switches are located on the pilot's overhead switch panel. Under normal operating conditions, all windshields and side panels will utilize anti-fogging continuously, except during anti-icing operation. Anti-fogging is turned on automatically when the anti-icing system is de-energized. Anti-fogging of the side panels and sliding windows is continuous when the ANTI-FOG switches are placed in the ON position. The anti-ice and anti-fog systems are controlled by three anti-ice and two anti-fog switches located on the windshield anti-ice and anti-fog section of the pilot's overhead control panel. All five switches have two positions, ON and OFF. The three anti-ice switches are labeled PILOT, CENTER, and COPILOT. The anti-fog switches are labeled PILOT and COPILOT.



Windshield Anti-Ice and Anti-Fog System
Figure 15-9

In the ON position, the anti-fog conductive coatings are energized and cause the control boxes to increase the temperature of the inner glass on all panels above the dew point. At the same time the inner layers of vinyl are maintained at a temperature of 27 to 38 degrees C (80 to 100 degrees F).

Upon receiving a signal from the ice detector system, the control boxes disconnect the anti-fog circuits of the windshield panels. At the same time, power is directed to the anti-ice circuits of the main windshield panels. The outside surface temperature of the panels is maintained at 1.1 degrees C (35 degrees F) or above.

When the ice detector cancels the signal to the control boxes, the anti-ice coatings are de-energized and power is restored to the anti-fog coatings of the windshield.

PITOT TUBE ANTI-ICING SYSTEM

The pitot system probes, located on the lower left and right sides of the fuselage are equipped with integral electric heaters for the prevention of ice accumulation. The heaters are controlled by two ON-OFF toggle switches located in the forward left corner of the pilot's overhead switch panel. Two pitot heater amber indicating lights are located above the switches and illuminate when their respective pitot probe heaters are energized (see Figure 15-10).

Pitot Heater Protection

The pitot anti-icing system is protected by the pilot's and copilot's pitot heater fuses.

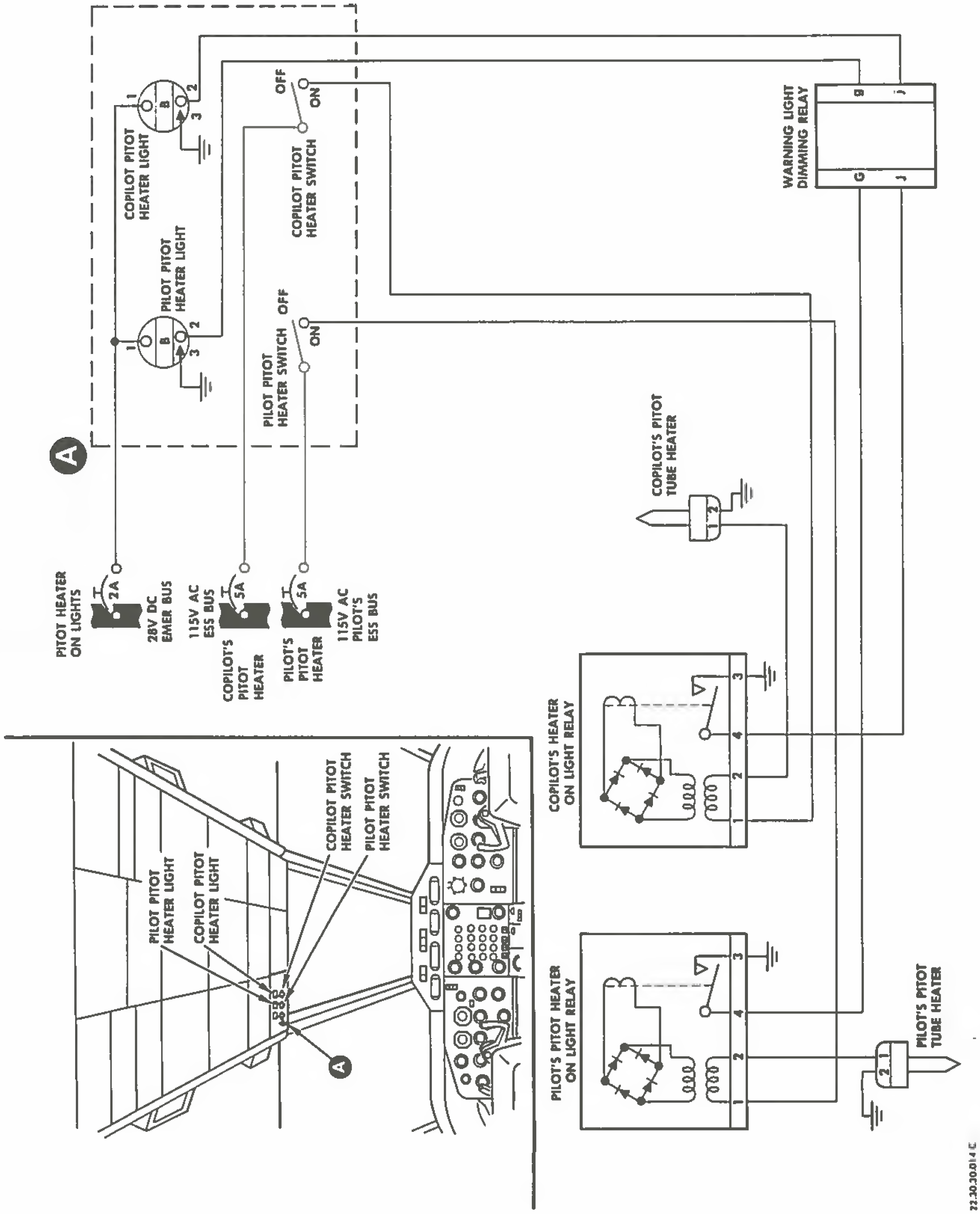
ENGINE ANTI-ICING SYSTEM

The engine air inlet duct lip, the engine compressor nose cone, and the inlet guide vanes on each engine are anti-iced by hot bleed air from the engines.

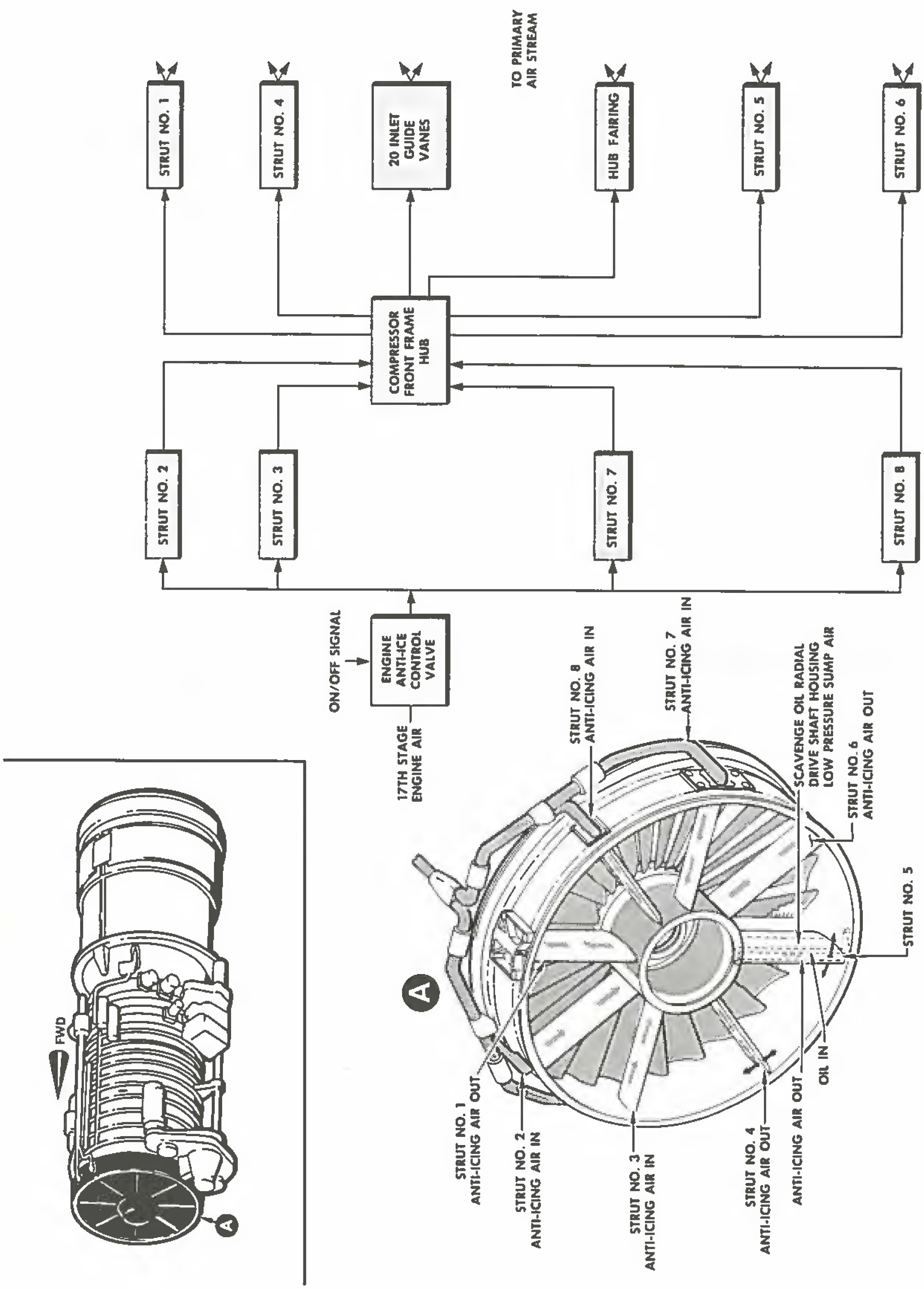
Engine Duct Lip Anti-Icing

Bleed air at 40 psig is directed from the engine bleed air pressure regulator-shutoff valve through a stub duct on the forward portion of the engine starter duct, into the nose cowl of each engine through a pressure regulating and shutoff valve. From this valve the air discharges forward into the duct lip distribution plenum chamber at a pressure of 11 psig (see Figure 15-11).

Air from the plenum chamber enters the passages between the double skin of the lip through 208 equally spaced 0.25-inch holes in the apex of the inner skin duct lip. It then passes aft through a baffle to a discharge plenum chamber where it is dumped into the primary engine air inlet through six equally spaced slots. A 2.5 psi differential is maintained in the double skin passages.



Pitot Heat Schematic
Figure 15-10



Engine Inlet Duct Lip Anti-Icing System
Figure 15-11

Engine Anti-Icing

The nose cone and inlet guide vanes of each engine are protected from ice by bleed air taken directly from the engine's bleed air manifold. The bleed air passes through an engine anti-ice pressure regulator-shutoff valve, reducing pressure to 20 psig, and is then circulated through passages in the guide vanes and nose cone. The engine anti-ice valves are controlled by the same switches that control the duct lip anti-ice valves.

Engine Anti-Ice Control Switches

For automatic operation of the engine anti-icing system, the four AUTO-OFF-MANUAL ON toggle switches in the engine anti-ice switch panel are placed in the AUTO position. This places the engine anti-ice valves under the control of the automatic ice detector system. If the automatic control and detection system should malfunction, the valves may be operated manually (see Figure 15-12).

Engine Anti-Ice Malfunction Indicators

Four amber warning lights directly above the control switches illuminate MAL FUNCT if their respective valves fail to operate properly under any required condition.

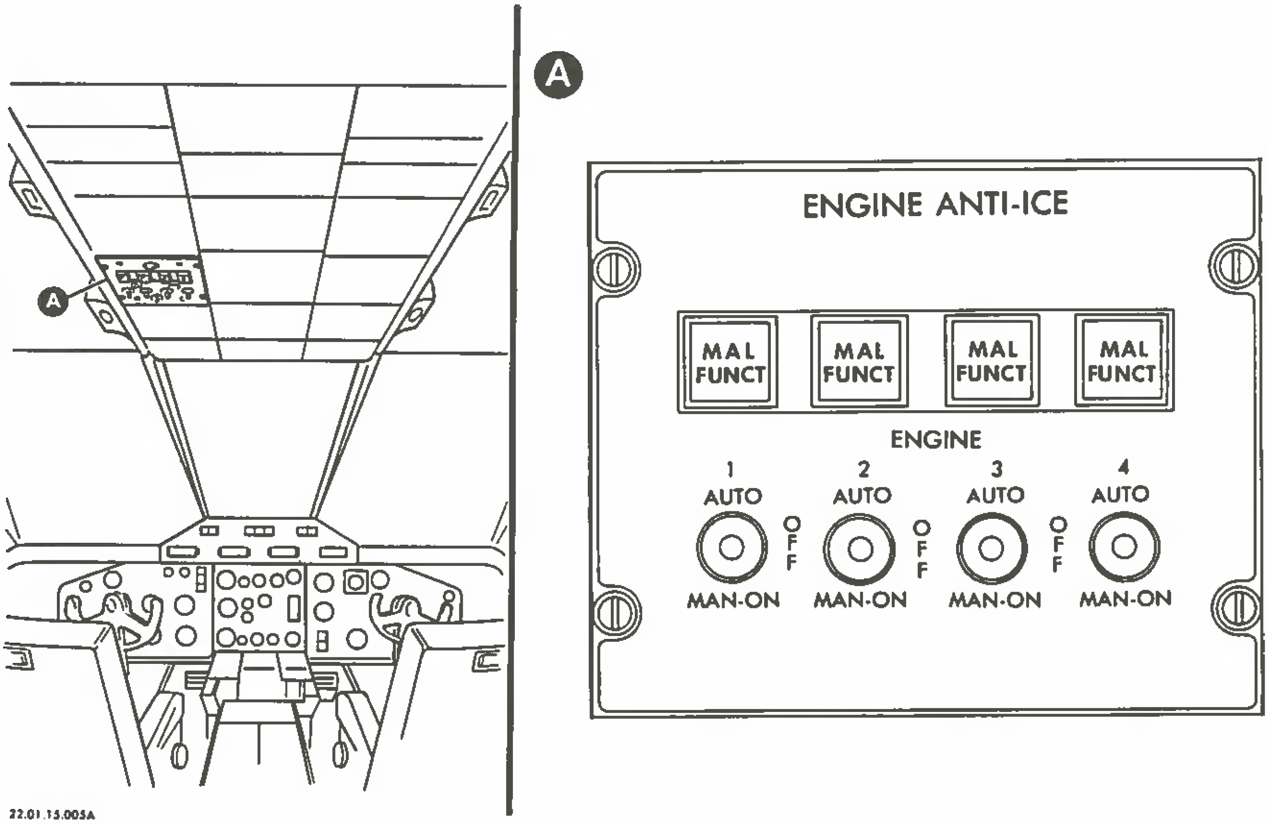
WINDSHIELD RAIN CLEARING SYSTEM

Proper visibility during rain conditions is assured by a bleed air rain clearing system. Bleed air is routed forward under the flight deck floor to the pilot's and copilot's windshields from the fuselage bleed air duct. The air passes through a rain clearing valve which also acts as the fuselage emergency bleed air isolation valve for that portion of fuselage ducting forward of the valve. From the rain clearing valve, the air continues through the duct to the sonic nozzles located at the lower inboard corner of the pilot's and copilot's windshields.

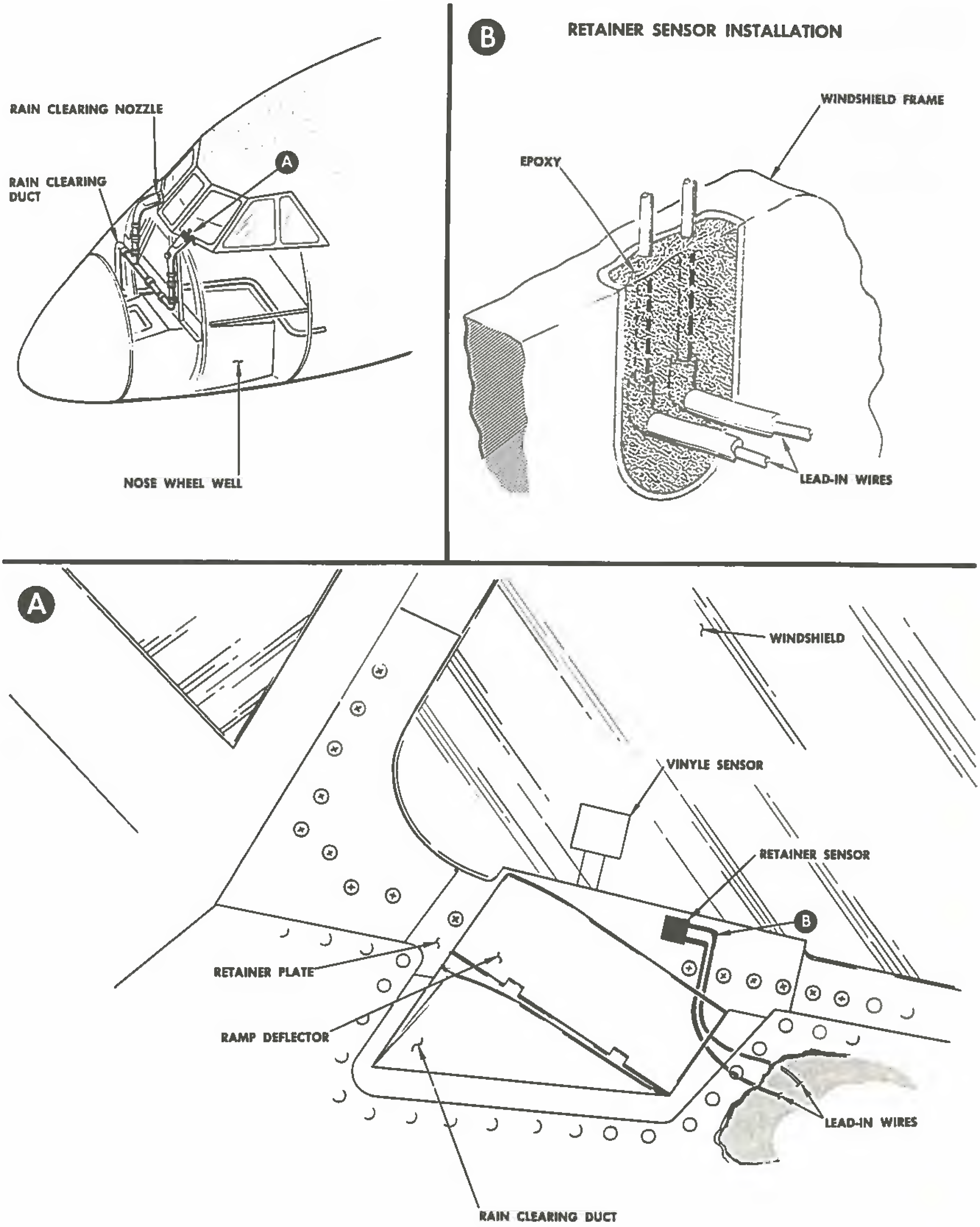
Rain Clearing Control Switch

The rain clearing system is controlled by an ON-OFF switch located on the pilot's overhead switch panel and by two temperature sensors located in the windshield panels (see Figure 15-13).

Placing the RAIN CLEAR switch in the ON position opens the rain clearing control valve. Regulated bleed air pressure at 40 psig is then ejected through sonic nozzles parallel with the airflow over the windshield panels. Because of the high air velocity across the surfaces, a barrier is formed which prevents rain impingement. Any rain that might actually penetrate the barrier will evaporate due to the heat of the windshield boundary layer.



Engine Anti-Ice Control Panel
Figure 15-12



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TEMPERATURE SENSOR LOCATIONS

Windshield Rain Clearing System
Figure 15-13

Overheat Protection

In case of an overheated condition, two temperature sensors located in the windshields are set to close the rain clearing valve automatically at 104 degrees C (220 degrees F). A blue indicator light above the rain clearing control switch on the overhead panel illuminates when bleed air is flowing through the rain clearing duct valve.

ADVERSE WEATHER AND BLEED AIR SYSTEMS POWER SOURCES

For detailed information, consult the WIRING DIAGRAM MANUAL.

AC Operated Equipment

Pilot's pitot heater
Engine duct lip anti-icing valves, engines No. 1 and No. 4
Ice detection probe heater, engine No. 1
Anti-fog coating, sliding and aft window panels
Anti-fog and anti-ice coating, left main windshield
Tail anti-ice heaters
Anti-fog and anti-ice coating, center and right windshields
Copilot's pitot heater
Ice Detection probe heater, engine No. 3
Bleed air overheat loop detectors, fuselage and wings
Skin overheat detectors, right and left wings
Space overheat detectors, right and left wings
Engine duct lip anti-icing valves, engines No. 2 and No. 3
Tail anti-ice heaters

28-volt dc Operated Equipment

Pitot heater indicating lights
Duct lip ice detector, engine No. 1
Bleed air overheat warning light, left wing
Bleed air regulator-shutoff valves, engines No. 1 and No. 2
Bleed air regulator-shutoff valve lights, engines No. 1 and No. 2
Bleed air regulator-shutoff valves, engines No. 2 and No. 4
Bleed air regulator-shutoff valve lights, engines No. 3 and No. 4
Bleed air high-pressure light
Duct lip ice detector, engine No. 3
Bleed air excess heat warning light, fuselage and right wing
Space overheat warning light
Anti-ice ON indicator light
Tail de-ice controller and indicator
Windshield rain clearing bleed air valve
Rain clearing temperature sensors

Section 16

OXYGEN SYSTEMS

TABLE OF CONTENTS

OXYGEN SYSTEMS

GENERAL OXYGEN SYSTEMS

Two high pressure, 1800 psi gaseous oxygen systems and four portable oxygen storage bottles are available for flight crew and passenger use. A diluter-demand type gaseous oxygen system supplies oxygen to the pilot, copilot and the flight engineer. The other gaseous oxygen system is a continuous flow type supplying oxygen to the passengers, stewardess' and the observer. The passenger oxygen system is automatic in operation when the cabin altitude exceeds 14,500 feet ± (1,000 feet). (see Figure 16-1). The four portable oxygen bottles are provided as therapeutic oxygen for passengers or the use of crew members.

FLIGHT CREW OXYGEN SYSTEM

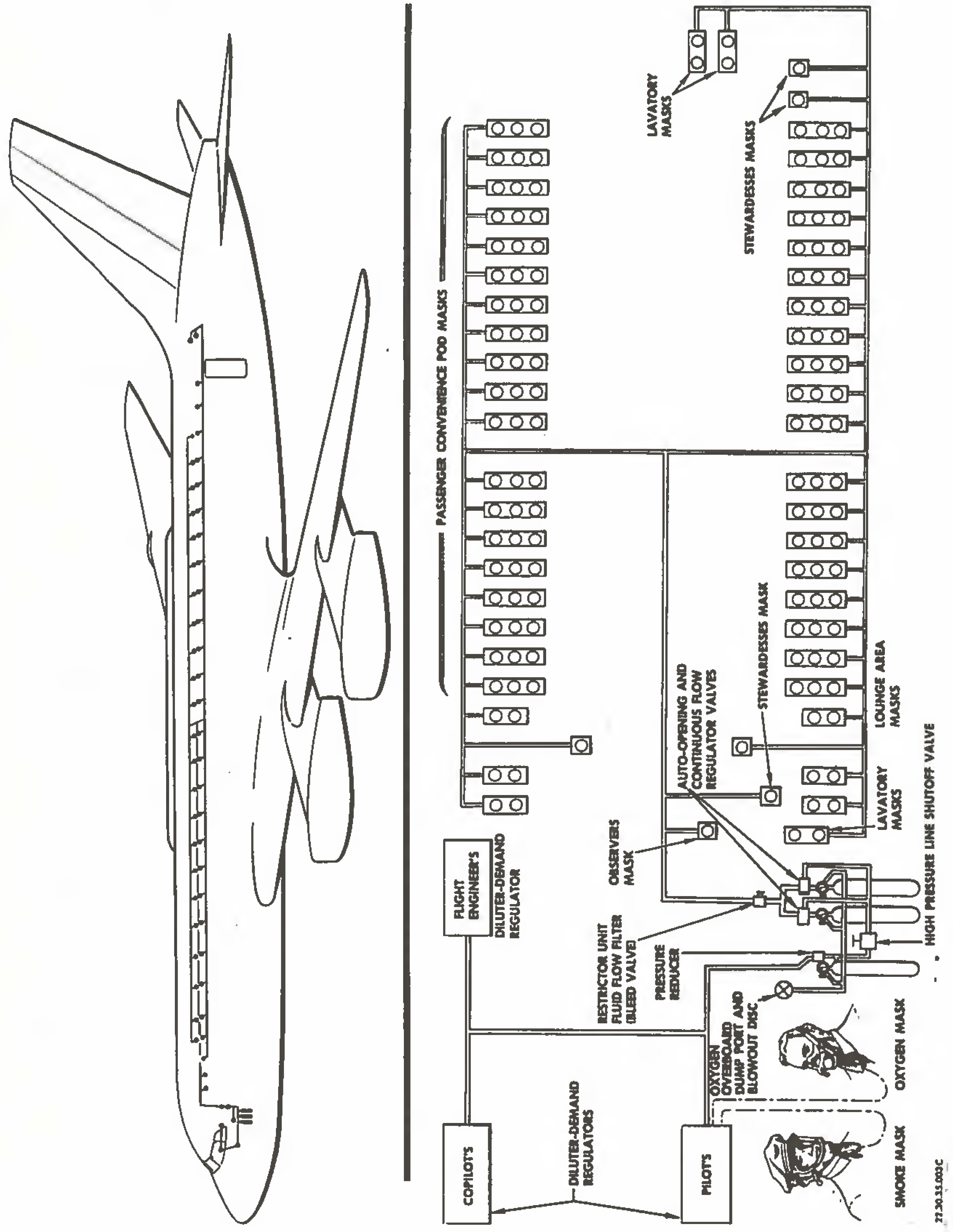
The flight crew oxygen system consists of an 1800 psi oxygen storage bottle, a system pressure reducer, three diluter-demand regulators, three mask selector valves, three normal oxygen masks and three smoke masks. A high pressure line shutoff valve is also included which permits the flight crew to divert oxygen from the passenger oxygen system storage bottles to the flight crew oxygen system.

Oxygen Flow Path

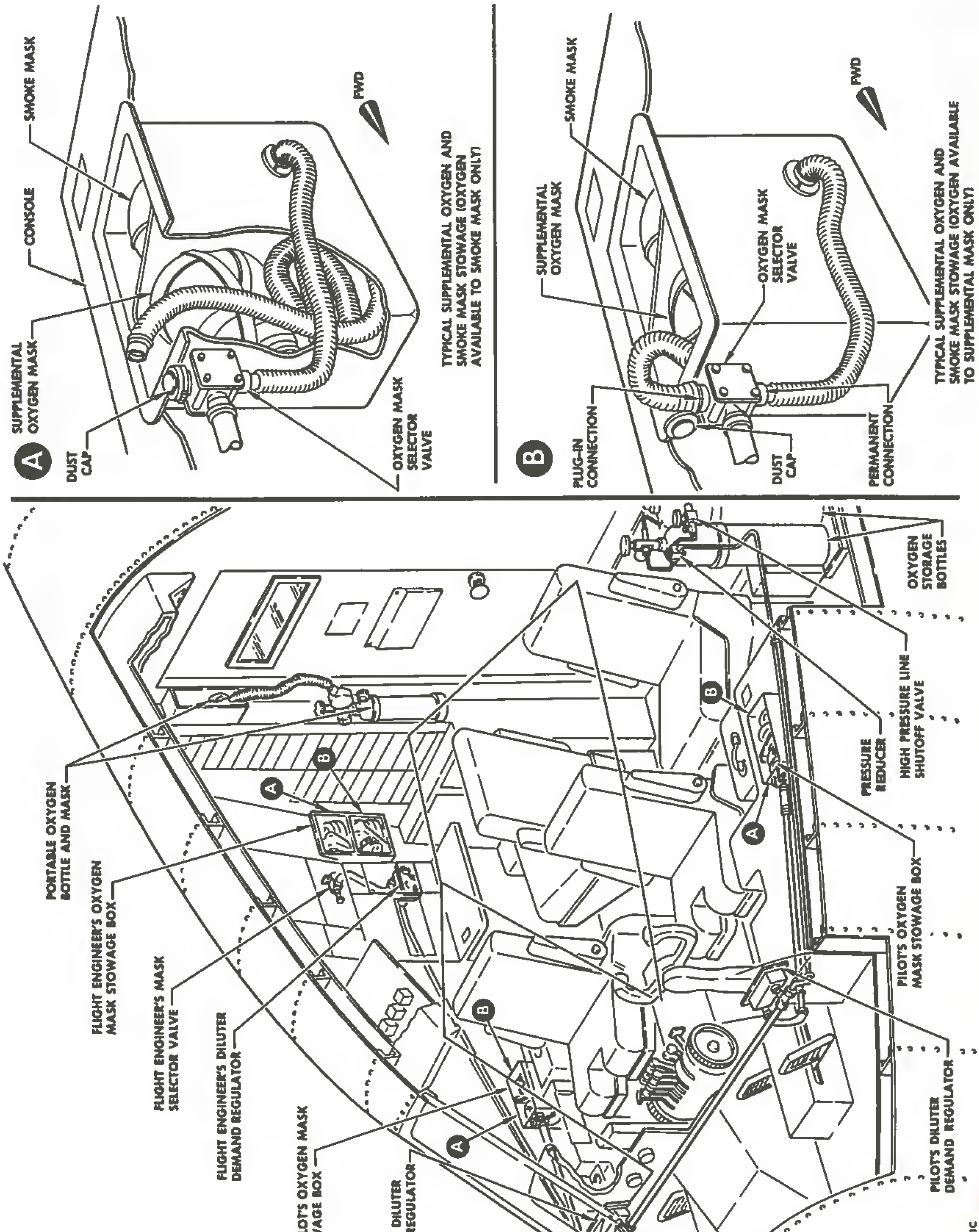
Oxygen flows from the source of supply through the pressure reducer where it is decreased in pressure from 1800 psi to 50-70 psig. From the reducer, oxygen flows to the pilot's, copilot's and flight engineer's diluter-demand regulators. With the regulator supply switches in the ON, NORMAL OXYGEN, and NORMAL positions, the regulator will supply a diluted amount of oxygen on demand up to 32,000 feet, at which point the regulator will be supplying 100 percent oxygen. With the switches in the ON, 100% OXYGEN, and NORMAL positions, the regulator's demand feature is eliminated and 100 percent oxygen is supplied at all times. With the switches in the ON, 100% OXYGEN, and EMERGENCY positions, the regulator will deliver an oxygen flow at a positive pressure of 3.5 inches of water. Switching from EMERGENCY to TEST MASK results in an oxygen flow under a positive pressure of 11.0 inches of water. The emergency position is used only when an added pressure of oxygen is required to flush the smoke mask after donning or when a positive pressure is necessary because of higher altitudes. The TEST MASK position is also used to check the regulator's delivery of oxygen.

Smoke Mask

The smoke mask is permanently connected to the mask selector valve. With the mask selector switch in the 100% OXYGEN position, and the normal oxygen mask unplugged, 100 percent oxygen is delivered to the smoke mask at a positive pressure of 3.5 inches of water (see Figure 16-2).



Oxygen System Schematic
Figure 16-1



Flight Compartment Oxygen Equipment
Figure 16-2

Oxygen Storage Bottles

The three oxygen storage bottles that furnish oxygen for the flight crew and passenger oxygen systems are mounted on the aft left side of the flight compartment. Provisions have been made for the installation of a fourth bottle. Each light-weight steel bottle has a capacity of 107.0 cubic feet of oxygen at 1800 psi. The contents of each bottle is indicated by a pressure gage mounted on the bottle. The gage is an integral part of the manually operated slow-opening valve that prevents pressure surges in the oxygen system.

Oxygen Storage Bottle Safety Disc

Each oxygen storage bottle is equipped with a built-in safety disc that will fracture at a bottle pressure of 2800 to 3000 psig, or at a temperature of 160 degrees Fahrenheit. When a safety disc is blown out, the entire contents of the bottle will dump overboard and eject a green blow-out disc mounted in the fuselage skin. The green blow-out disc provides a visual indication from the exterior of the airplane as to the condition of the oxygen bottle safety disc. The blow-out disc is located in the skin directly outboard of the oxygen storage bottles.

High Pressure Line Shutoff Valves

The high pressure line shutoff valve is located in the interconnecting line between the two passenger oxygen storage bottles and the forward flight crew oxygen storage bottle. The purpose of this valve is to allow the flight crew to use the passenger oxygen supply when necessary.

PASSENGER OXYGEN SYSTEM

The passenger oxygen system is the continuous-flow type supplied from two oxygen storage bottles in the flight compartment. Oxygen from each storage bottle flows through an auto-opening continuous flow regulator valve. The flow from the two valves is combined into one single line into the passenger compartment area where the line tees into two lines connecting to the overhead convenience pods on each side of the airplane. At each pod, the lines connect to a rotary valve for each mask and to a pneumatic latch. Plastic tubes connect the rotary valves to the passenger face masks.

Automatic Actuating System

The two auto-opening continuous flow regulator valves are controlled by barometric pressure and are adjusted to open at a pre-determined altitude. The regulators reduce the 1800 psig supply oxygen to a pressure of 13-55 psig. When oxygen flow occurs, the pneumatic latches are released and the passenger oxygen masks are dropped. A manual override toggle switch at each valve allows oxygen to be released to the passenger masks manually. However, between cabin altitudes of 10,000 feet and 14,500 feet (±1,000 feet), the masks must be manually removed from their stowage positions.

Passenger Oxygen Masks

The number of masks in the passenger oxygen system, and their locations, varies with different interior configurations of the airplane. Each mask assembly consists of a face mask, various valves, a reservoir bag and connecting tubing. One additional mask for each seat row is provided to supply oxygen for children in arms. Each lavatory also has available two automatically released oxygen masks.

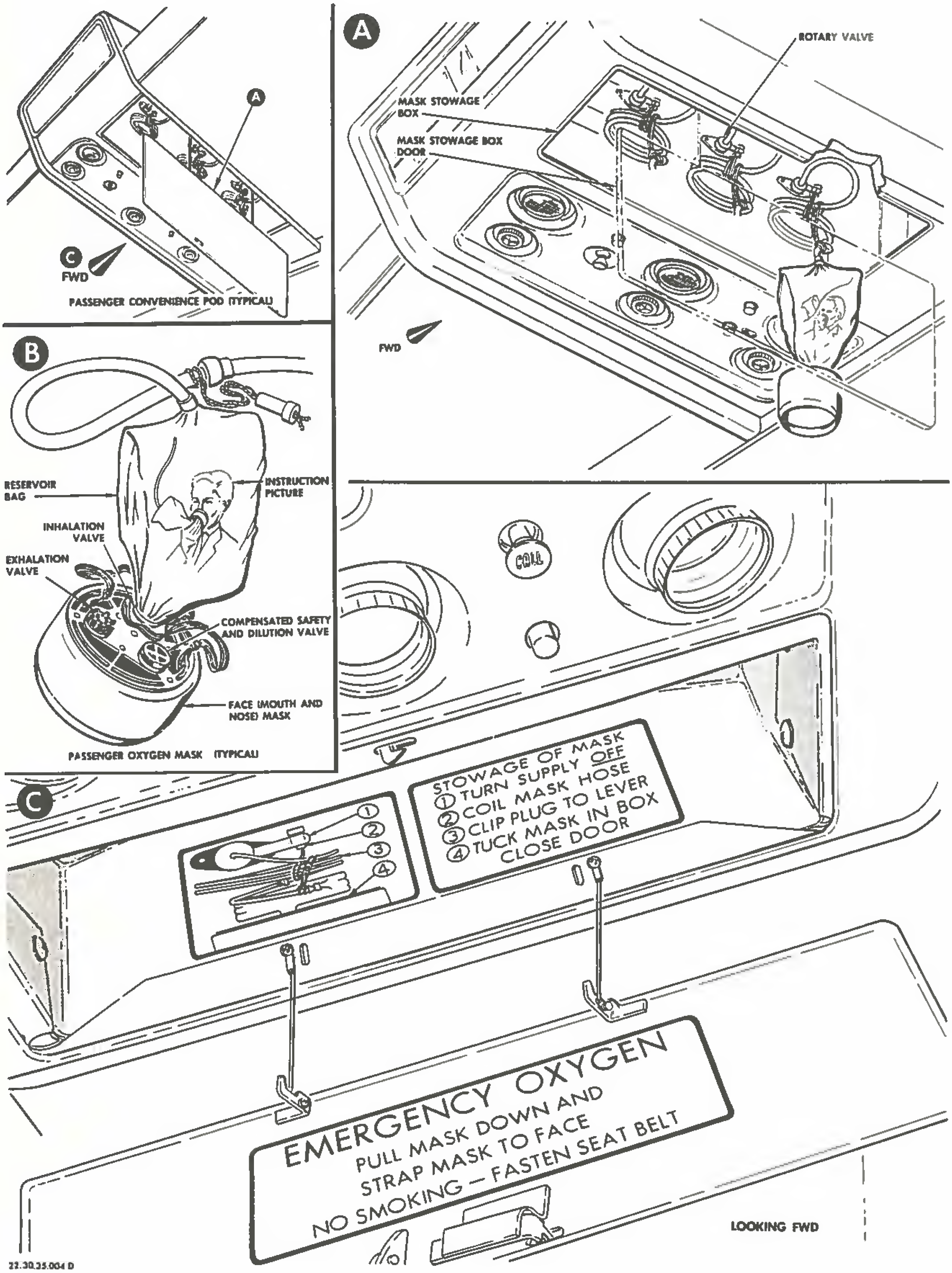
The passenger's, stewardess's and observer's oxygen masks are supplied with oxygen by manually pulling the masks downward after they have been released from the mask stowage panel. The downward pull actuates the rotary valves that permit oxygen to flow into the face masks (see Figure 16-3).

PORTABLE OXYGEN BOTTLES

One portable high pressure oxygen bottle, regulator and mask is located in the flight compartment.

Three portable high pressure oxygen bottles, with demand regulators and continuous flow masks are located in the passenger compartment as follows:

1. One in the forward left hand hatrack stowage bin.
2. One in the left hand mid-cabin stowage bin.
3. One in the aft left hand stowage bin.



Passenger Oxygen Mask Stowage
Figure 16-3

Section 17

FIRE PROTECTION

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FIRE PROTECTION

GENERAL FIRE PREVENTION

In order to prevent any possible spread of a fire in the engine areas, the pod and pylon of each engine installation is divided into four separate compartments as follows: (see Figure 17-1)

1. The compressor and accessory section is divided into a forward compartment.
2. The combustion liner and turbine sections are divided into an aft compartment.
3. The pylon is isolated from the pod by a liquid and vapor tight titanium firewall capable of resisting a 2000 degree F flame for a period of 15 minutes.
4. The pylons are isolated from the under surface of the wing by means of a draft seal.

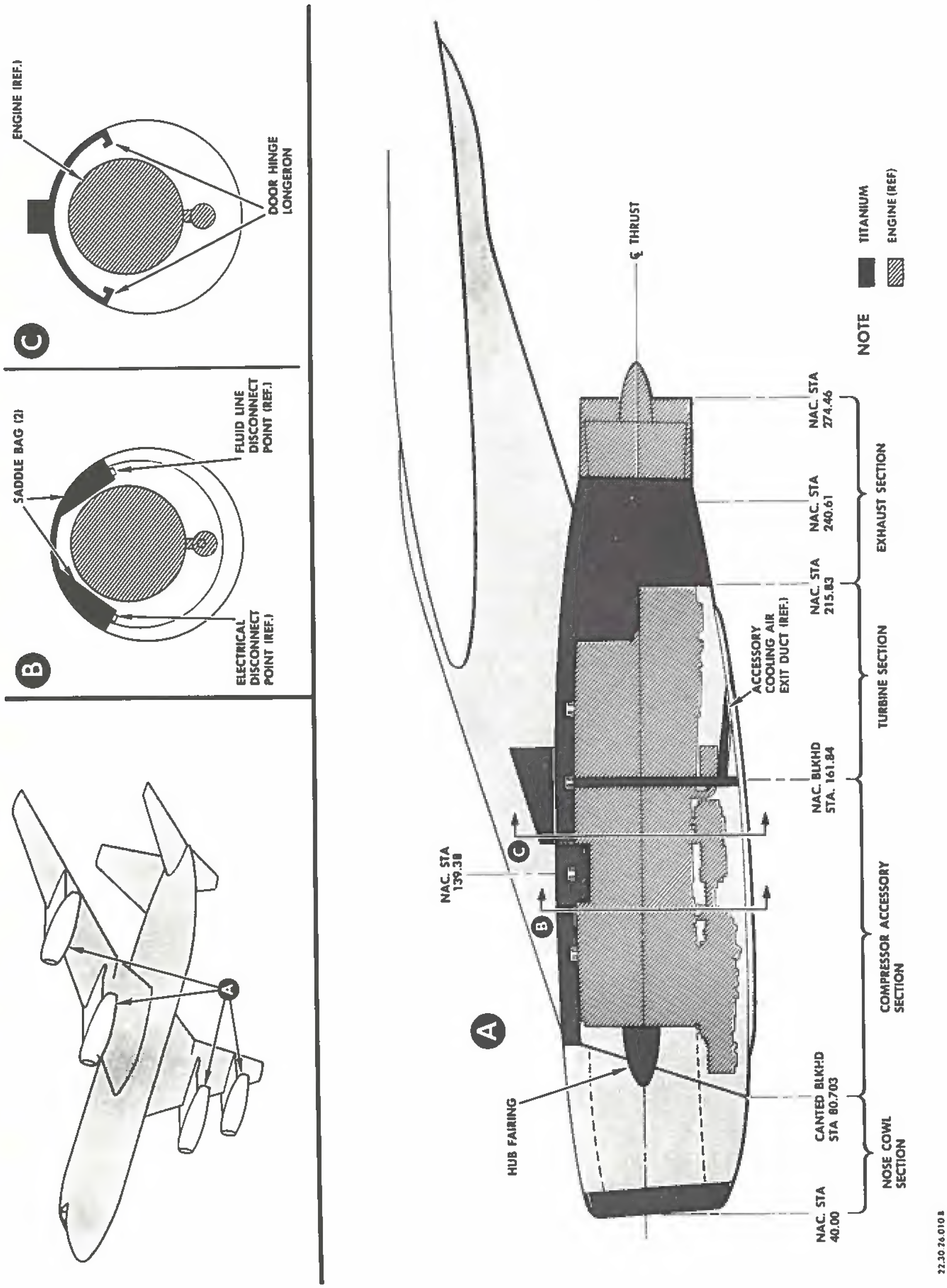
FIRE PROTECTION SYSTEM

The fire protection system consists of two subsystems: fire detection and fire extinguishing (see Figure 17-2). Continuous loops in the engine areas detect an overheat or fire condition, and transmit a signal through control units and power relays to a warning bell and warning lights in the flight compartment. A FIRE PULL "T" handle is then pulled to shut off the supply of fluids to the engine in which the fire exists. Pulling the fire-pull handle also exposes a previously inaccessible extinguishing agent release switch. When actuated, this switch releases the extinguishing agent to the affected area. If a fire has been extinguished and then re-occurs, a two-shot configuration makes it possible to direct the contents of another container to the area (see Figure 17-3).

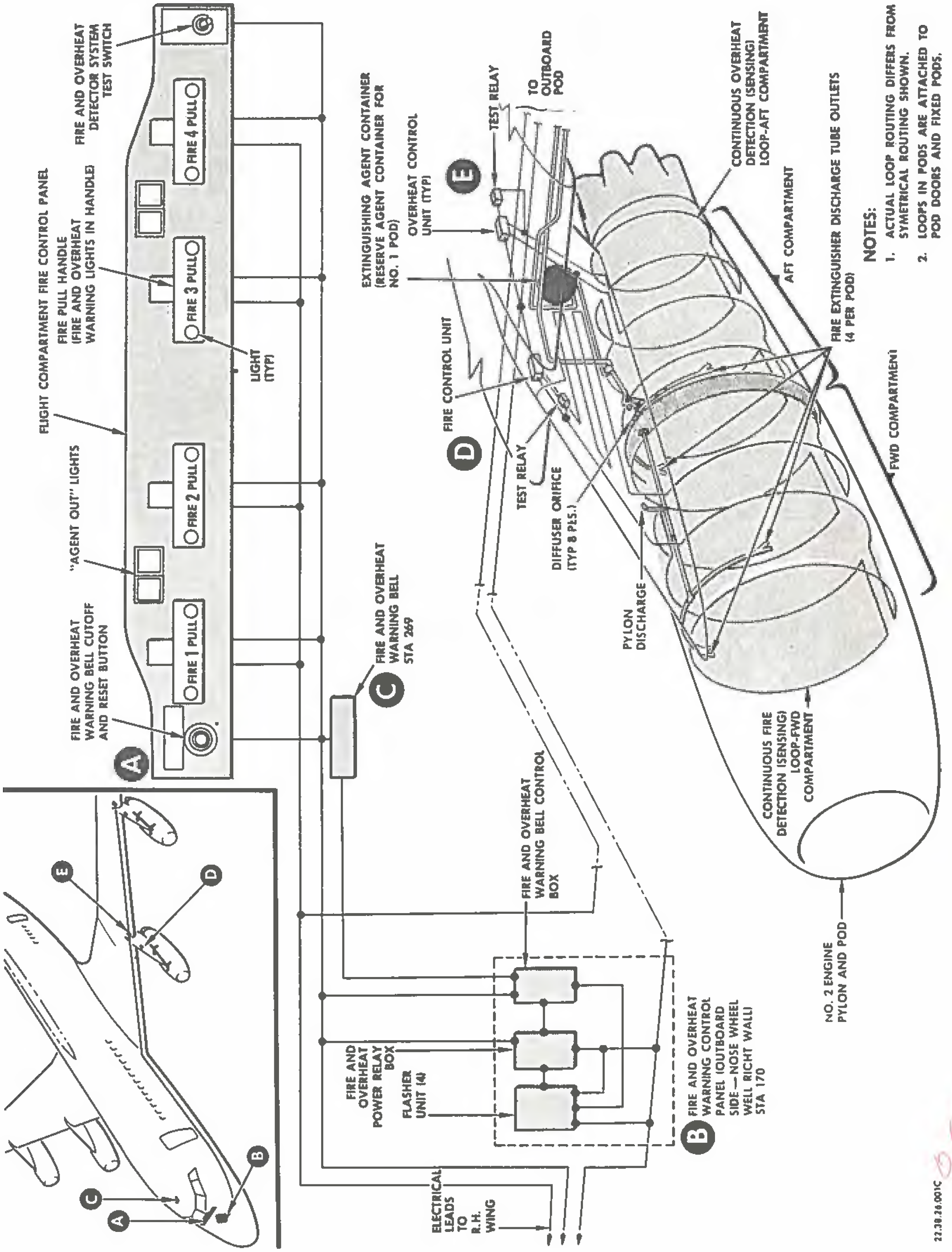
FIRE DETECTION SYSTEM

Each engine pod and pylon contains a fire detection circuit. The overheat and fire detection continuous loop is routed in a "maximum hazard" pattern around the burner and tailpipe portion of the engine in the aft pod compartment. A second fire detection loop is routed in the same manner around the compressor and accessory portions of the engine in the forward pod compartment and also extends into the pylon.

Whenever any part of the continuous loop circuit in the forward compartment is exposed to fire temperature, the detector warning lights, located in the fire-pull handle, illuminate and the fire warning bell will sound. When any part of the continuous loop circuit in the aft pod compartment is exposed to overheat or fire temperature, the detector warning lights, located in the fire-pull handle, flash intermittently and the warning bell sounds continuously.



Nacelle Construction for Fire Prevention
Figure 17-1



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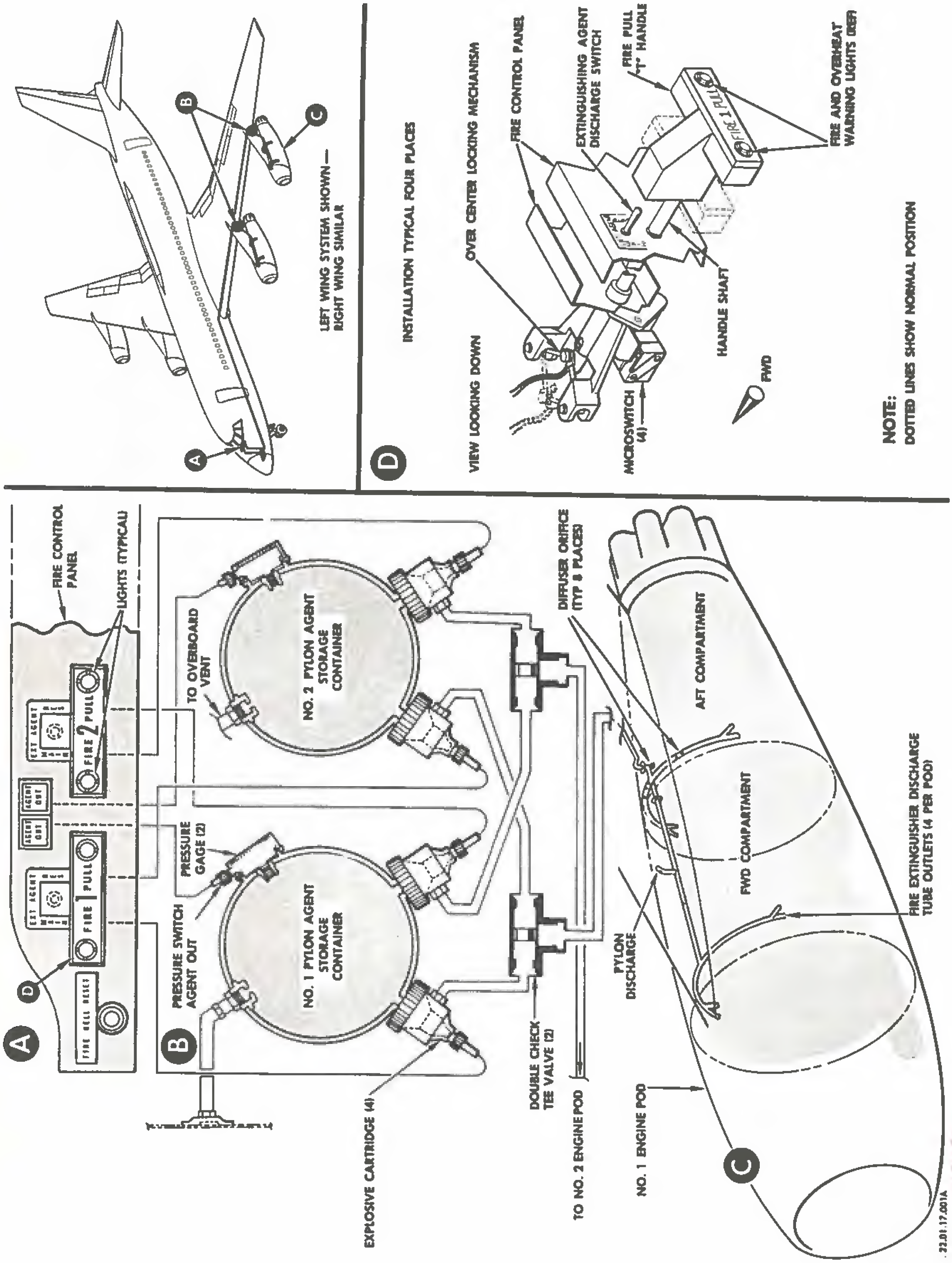
Fire Protection System
Figure 17-2



Fire Extinguishing System Schematic
Figure 17-3

Steady Illumination

Steady illumination of any of the four fire-pull "T" handle warning lights and steady sounding of the fire warning bell should be assumed by the flight crew to be an indication of engine fire and the fire fighting procedure should be performed immediately.

Flashing Illumination

Flashing illumination of any of the four fire-pull "T" handle warning lights and continuous sounding of the fire warning bell should first be assumed to be caused by an excessive aft compartment temperature. Retard the appropriate power control lever and observe if the warning signals cease. If not, assume an actual fire exists and follow the fire fighting procedure.

FIRE EXTINGUISHING SYSTEM

The fire extinguishing portion of the fire protection system includes four spherical containers for storage of the extinguishing agent. One container is located in each pylon, just forward of the fuel system refueling panels in the inboard pylons and in the same relative position in the outboard pylons. Two discharge valves operated by electrically discharged cartridges, a main and reserve control and corresponding agent outlet tubing, attached to each container, release and route the extinguishing agent either to the corresponding pod and pylon, or when the reserve configuration is selected, to the other pod and pylon on the same wing. This two-shot, crossfeed configuration makes it possible to release a second charge of fire extinguishing agent to the same pod and pylon if another fire breaks out, without having two containers for each engine.

Fire Extinguishing Agent

The fire extinguishing agent utilized in Bromotrifluoromethane (CF_3Br) and each container is charged with 6.55 lbs of the agent.

Discharge Rate

Each container will rapidly discharge so as to provide a concentration of 15 percent by volume throughout the compartments for a duration of 1/2 second. This concentration occurs within 3 to 4 seconds.

FIRE EXTINGUISHING SYSTEM CONTROLS

The fire extinguishing system controls consist of four fire-pull "T" handles, four agent-out indicator lights, and four switches to select main or reserve release of the fire extinguishing agent. A test switch is provided for test of the fire and overheat system. A fire warning reset button is also provided. All controls are on the fire control panel located in the center of the pilots' instrument panel anti-glare shield.

Fire-Pull "T" Handles

Four FIRE-PULL "T" handles are provided, one for each engine area. Each handle contains the fire detection system warning lights. Pulling the "T" handle exposes a previously inaccessible extinguishing agent switch and also actuates microswitches which route 28-volt dc power directly to the emergency fuel shutoff valve, engine bleed air shutoff valve, and the hydraulic suction shutoff valve. Also, hydraulic low pressure warning lights are disarmed.

Extinguishing Agent Switches

Pulling the "T" handle exposes the EXTINGUISHING AGENT three position spring-loaded switches. Actuating the switch momentarily to MAIN releases the extinguishing agent for the pod and pylon involved. Actuating the switch momentarily to RESERVE releases the extinguishing agent from the adjacent "same wing" engine fire protection system to the pod and pylon involved.

Extinguishing Agent Out Lights

A pressure sensitive switch in each container illuminates the AGENT OUT light when the contents of the container have been released or pressure should leak below 250 psi. Movement of the extinguishing agent switch to the reserve position illuminates the AGENT OUT light of the adjacent engine fire extinguishing agent container when its contents have been released.

Fire and Overheat Test Switch

A three position TEST switch is located on the right side of the fire control panel for pre-flight test of the fire detection system. Moving the switch to FIRE position illuminates the steady burning fire warning lights, and sounds the fire warning bell, if the circuit is operating properly. Moving the switch to O'HEAT position illuminates the flashing overheat warning lights, and activates the fire warning bell, if the circuit is operating properly.

Fire Bell Reset Switch

A momentary push-button switch is provided on the left side of the fire control panel to reset the fire warning bell after it has been activated.

EXTINGUISHING AGENT STORAGE CONTAINER PRESSURE INDICATORS

Each extinguishing agent storage container is provided with a pressure gage marked in 50 psi increments form 0 to 1500 psi. Storage containers are normally pressurized to 600 lbs at 21 degrees C (70 degrees F) with nitrogen used as the pressurizing media.

GROUND FIRE FIGHTING PROVISIONS

Two fire access doors are installed in the left side of each engine nacelle. These doors are clearly marked, pop-in types, and are faired to the skin of the nacelles when not in use. Portable fire extinguishers, usually the cone type, can be thrust against one or both of the doors for ground fire extinguishing purposes (see Figure 17-4).

WARNING: CONSIDERABLE CARE SHOULD BE TAKEN BY GROUND PERSONNEL IN GROUND FIRE EXTINGUISHING PROCEDURES TO AVOID DEATH OR INJURY FROM OPERATING JET ENGINE INTAKES OR EXHAUSTS.

HAND FIRE EXTINGUISHING EQUIPMENT WEIGHT AND CAPACITIES

Three winterized hand water fire extinguishers are provided: one at the forward main entrance area and two on the forward side of the partition immediately forward of the aft entrance. A hand type CO_2 extinguisher is located in the flight compartment near the circuit breaker panel.

EMERGENCY FIRE AXE

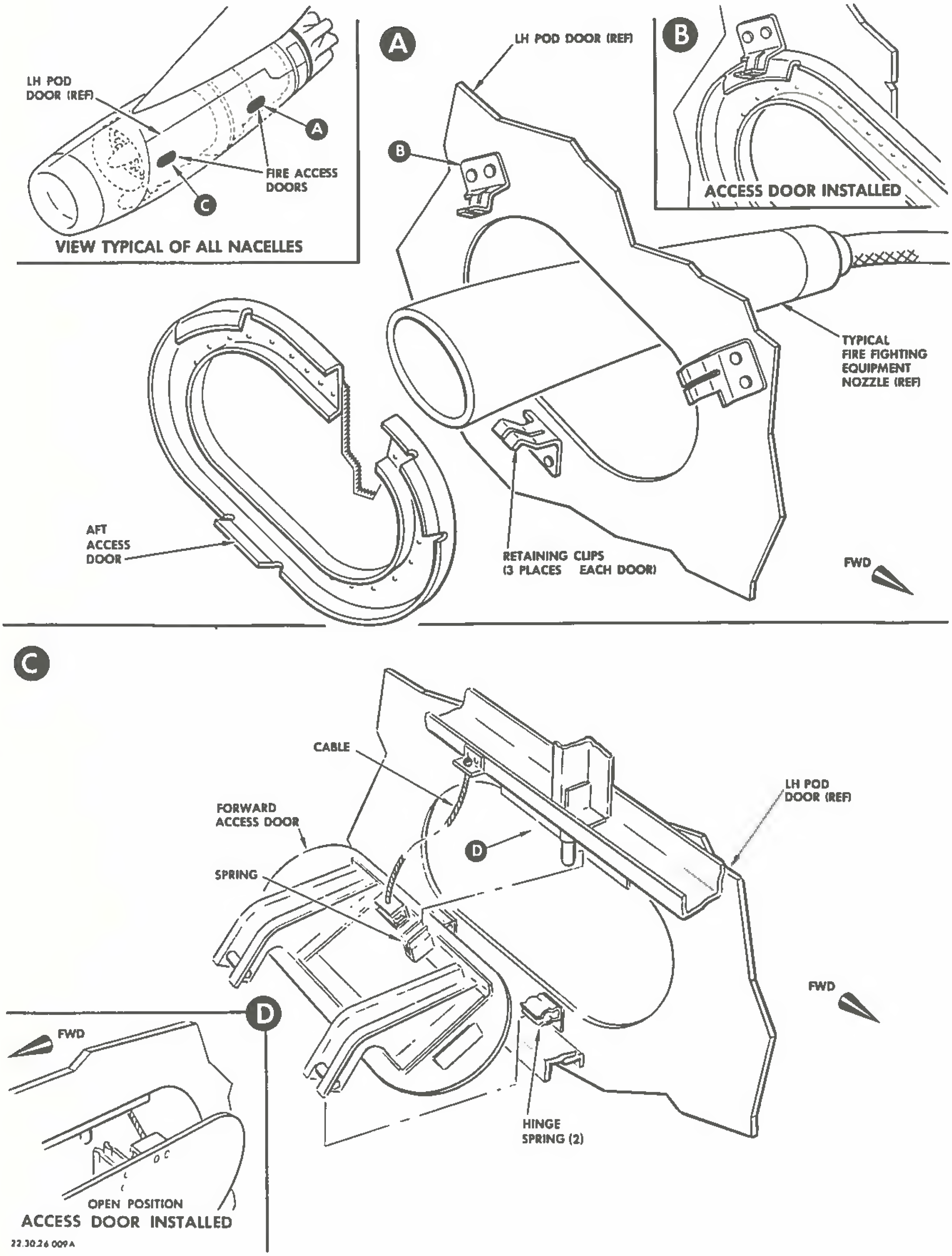
An emergency fire axe is provided in the flight compartment, mounted in a holder on the flight compartment door.

SMOKE MASKS

Smoke masks are provided for the captain, first officer, flight engineer, and observer. The smoke masks connect to the airplanes flight crew oxygen system.

FIRE PROTECTION SYSTEM ELECTRICAL SOURCES

Fire detection system pilot's essential ac bus and the 28-volt dc emergency bus. Fire extinguishing system (including all shutoff valve electrical power, fire and overheat warning lights, and bell), 28-volt dc emergency bus.



Ground Fire Fighting Access Door
Figure 17-4

Section 18

FLIGHT AND NAVIGATION INSTRUMENTS

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FLIGHT AND NAVIGATION INSTRUMENTS

GENERAL DESCRIPTION - FLIGHT AND NAVIGATION SYSTEMS

The pilot's and copilot's instrument panels each displays a full set of flight and navigation instruments. The flight instruments present altitude, attitude, and airspeed information. The navigation system includes magnetic and radio equipment for indicating heading, bearing, and position information. The flight crew is equipped to take advantage of the aids to air navigation provided by the federal airways system.

Some of the flight instruments are refinements of standard types that have been in common use for many years while others represent new concepts or combinations of two or more types of information in a single instrument.

Those flight instruments which indicate airspeed and altitude are combined in a Kollsman Integrated Flight Instrument System (KIFIS) which automatically compensates for the errors inherent in instruments that depend on air pressure for their operation. Stable directional information is provided by a Polar Path Compass System which combines gyro stability with the directional properties of the magnetic compass. VOR/ILS, heading, and attitude information and computed attitude commands are displayed in two instruments in the Bendix Flight Director System. Radio bearing and geographic position information is provided by the Automatic Direction Finding and the Marker Beacon Systems.

Search-weather radar furnishes information for avoiding areas of severe turbulance. A terrain-clearance radar system is also provided.

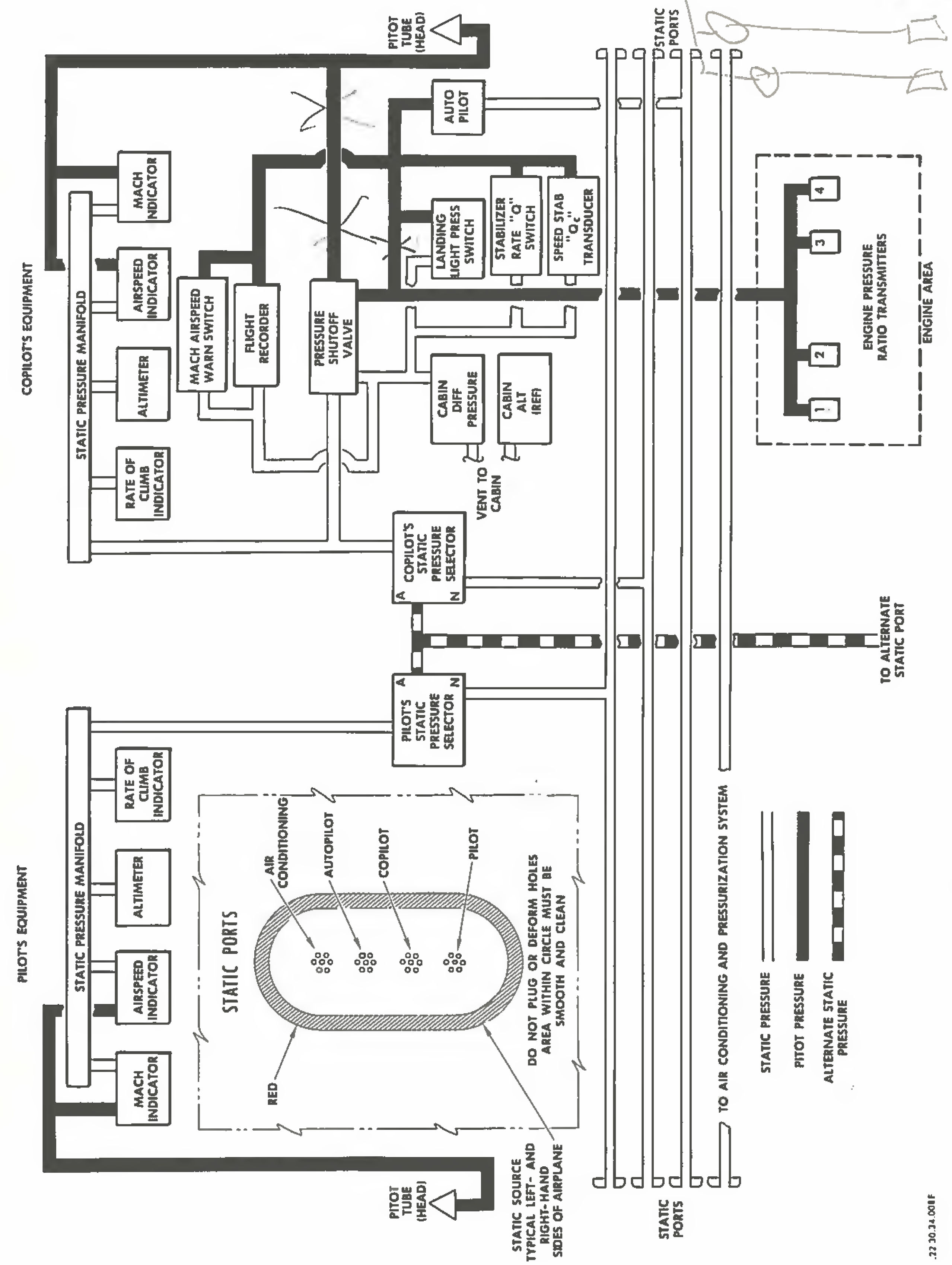
A transponder beacon system is included to provide a means of radar identification to expedite air traffic control in congested areas.

PITOT-STATIC SYSTEM

Most of the flight instruments and several component systems of the airplane depend upon static or dynamic pressure or both for their operation. The pitot-static system provides a continuous source of dynamic and static pressure for the flight instrument, flight recorder, pressure ratio transmitters, pressure switches, autopilot, air conditioning and pressurization system, and cabin pressure indicators. The system consists of static ports, pitot heads, shutoff and selector valves, and the necessary tubing and connections (see Figure 18-1).

Pitot System

The pitot system supplies dynamic or pitot air pressure to the pilot's and copilot's instruments, the autopilot, the flight recorder, and landing light pressure switch, and the engine pressure-ratio transmitters. The two pitot heads, one on each side of the fuselage, are equipped with integral electric heaters to prevent icing. The heaters are operated by two toggle switches on the pilot's overhead switch panel.



Pitot-Static System Block Diagram
Figure 18-1

The pitot head on the left side of the fuselage supplies pitot pressure to the pilot's machmeter and airspeed indicator. The right pitot head supplies pitot pressure to the copilot's machmeter and airspeed indicator. The copilot's pitot line also connects through a 2-position shutoff valve with the autopilot, the landing light pressure switch, and the engine pressure-ratio transmitters, the flight recorders, and mach/airspeed warning switch. This valve normally ON, is closed only if a leak develops in one of the pitot lines downstream from the valve.

Static System

The static system includes eight normal ports, four on each side of the fuselage, and an alternate port located in the unpressurized area in the tail cone. The eight normal ports are paired, one on each side of the fuselage, to supply static pressure to the pilot's instruments, the copilot's instruments, the autopilot, and the air conditioning and pressurization system.

The pilot's static system includes one pair of the normal ports, the alternate static port, a selector valve, and a static pressure manifold. The pilot's machmeter, airspeed indicator, rate-of-climb indicator, and altimeter are connected to the manifold. A NORMAL-ALTERNATE selector valve on the left side of the pilot's instrument panel connects the pilot's instruments with either normal or alternate static pressure.

The copilot's static system includes a second pair of normal static ports, the alternate static port, a selector valve, and another static pressure manifold. The copilot's machmeter, rate-of-climb indicator, airspeed indicator, and altimeter are connected to this manifold. Static pressure from the copilot's static pressure manifold is also supplied to the landing light pressure switch, the flight recorders, the mach/airspeed warning switch, and the cabin differential pressure gage through a 2-position ON-OFF valve. A NORMAL-ALTERNATE selector valve on the right side of the copilot's instrument panel connects the copilot's instruments with either normal or static pressure.

Static System Error Correctors

The static system of a moving airplane tends to introduce errors into the indications of some of the flight instruments. For all practical purposes, these errors may be assumed to be a function of Mach number and angle of attack. Two static system error correctors, included in the KIFIS system control chassis, accomplish the required error corrections for each specific condition through 3-dimensional cams incorporated in each unit. Two correction cams are provided for each altimeter, one for the normal static source, the other for the alternate static source. A servo rotates the cam in response to a signal from the machmeter and a cam follower moves radially across the cam surface as a function of angle of attack. The combined inputs, resolved by the follower, produce a synchro signal proportional to the required system error correction.

KOLLSMAN INTEGRATED FLIGHT INSTRUMENT SYSTEMS (KIFIS)

The Kollsman Integrated Flight Instrument System, also known as the Air Data System, provides continuous indications of indicated and true airspeed, Mach number, altitude, and static air temperature. The system includes computer units which correct for variations in angle of attack, altitude, airspeed, and temperature. The computer units are located in the control chassis assembly in the electronics compartment. The system has an operating range from sea level to 50,000 feet with temperatures ranging from -30 degrees C to + 50 degrees C (see Figure 18-2). The system also supplies information for the Speed Stability Augmentation (Mach Trim) System.

The instruments included in the KIFIS are: airspeed indicators, machmeters, altimeters, a true airspeed indicator, and an outside air temperature indicator. Switches to test the system are also included.

CAUTION: THE AIR DATA SYSTEM IS POWERED BY 115-VOLT, 400 CPS, AC CURRENT. WITH LOSS OF ELECTRIC POWER, THE INSTRUMENTS WILL OPERATE CONVENTIONALLY BUT THE RESULTANT INDICATIONS WILL BE UNCORRECTED FOR OPERATING VARIABLES.

Airspeed Indicator

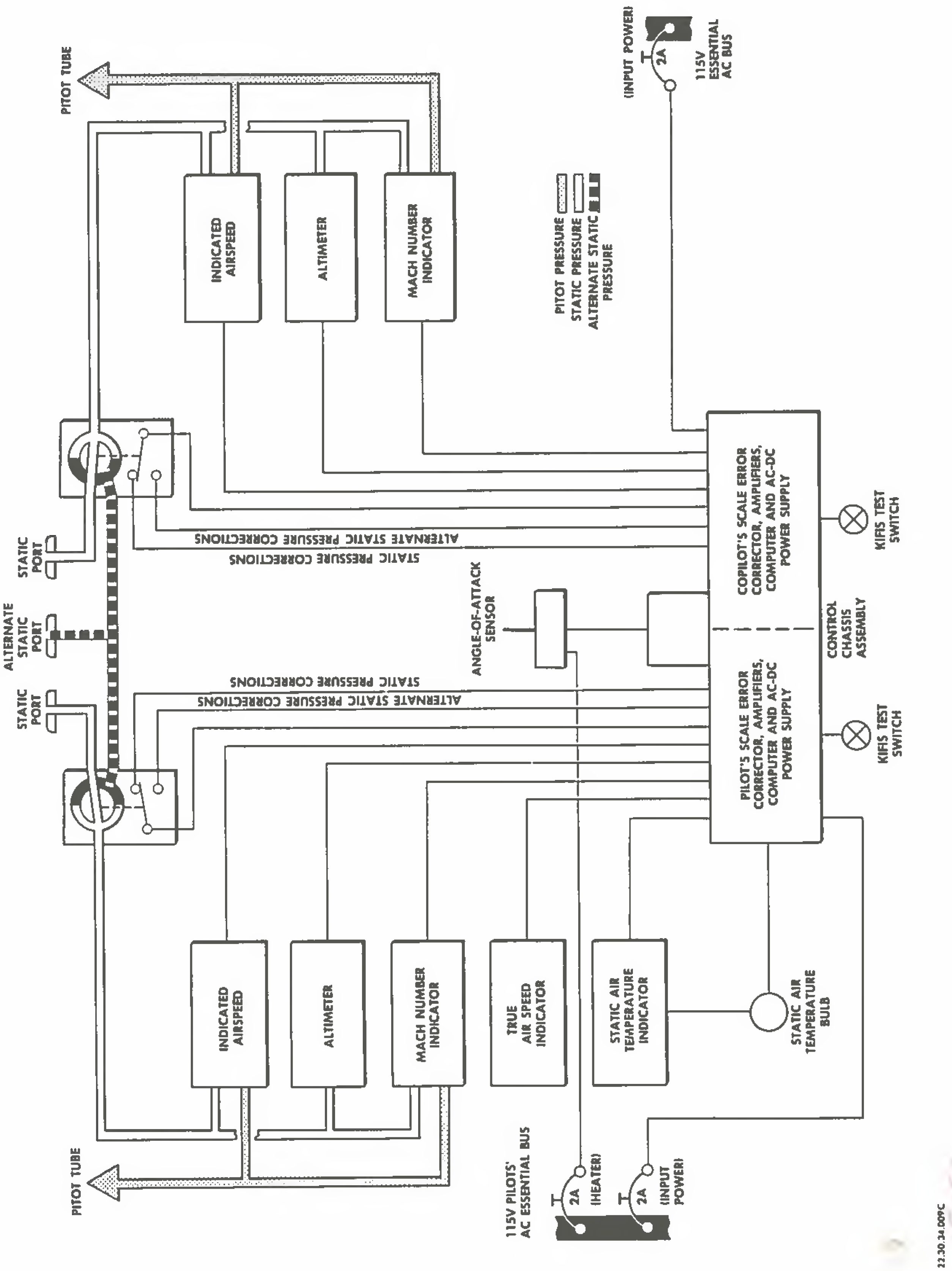
The pilot's and copilot's airspeed indicators are two-in-one instruments providing a continuous indication of indicated airspeed and maximum allowable airspeed. Each instrument includes an airspeed dial calibrated in knots, a white pointer to indicate airspeed, and a black pointer with red stripes to indicate the maximum allowable airspeed under all operating altitude and temperature conditions.

The airspeed mechanism consists of a pressure differential diaphragm which compares the static and pitot pressures and positions the white pointer to read indicated airspeed against the calibrations on the airspeed dial. The red striped black pointer is positioned with respect to the same dial by a static pressure diaphragm and indicates the maximum allowable airspeed.

The range of the airspeed indicator is 450 knots. The maximum allowable airspeed pointer is operable over the full range of the instrument. A warning bell sounds if the indicated airspeed exceeds the maximum allowable airspeed. The bell circuit can be tested by a test button located on the pilot's flight instrument panel.

Angle of Attack Sensor

The angle of attack sensor is mounted on the left side of the fuselage near station 473. The unit consists of a probe with two horizontal slots exposed to the ram air, a pressure differential diaphragm, and dual transmitting synchros. The slots sense the direction of the relative wind in relation to the attitude of the airplane and the synchros provide an angle of attack signal to the computer units. An integral heating unit in the sensor automatically maintains an adequate temperature to prevent icing.



Kollsman Integrated Flight
Instrument System Diagram
Figure 18-2

Altimeters

A separate altimeter is provided for the pilot and copilot. Each instrument is the drum-pointer type with a range of -1000 feet to +50,000 feet. A radial pointer indicates hundreds of feet. Thousands of feet are indicated by a rotating drum visible through an opening in the face of the instrument. Two cutouts in the face of the instrument expose additional drums indicating barometric pressure in inches of mercury and in millibars. The barometric drums are positioned by a knob at the lower left corner of the case.

The pressure sensing mechanism of the altimeter incorporates a self-balancing dual static pressure diaphragm system which provides a high-torque precision output with minimum friction. A servo mechanism compensates for scale and static pressure errors.

Altimeter Scale Error Correctors

A scale error corrector provides a correction for scale or calibration errors inherent in each individual altimeter. Each correction unit contains an adjustable cam that is rotated as a function of altitude. Scale error corrections generated by cam action are transmitted to a differential synchro which combines the scale correction signal with a correction signal from the static system error corrector. The combined signal is transmitted to the servo mechanism in the altimeter which applies the appropriate correction to the altitude indication.

An automatic spring return device within the altimeter removes all error corrections in case of a power failure and a chevron appears on the altimeter dial. The instrument continues to operate as a conventional altimeter.

With the altimeter functioning as an integral part of the KIFIS and receiving correction signals for scale and static system errors, the total altimeter errors are reduced to the following:

Altitude (Feet)	Total Error (± Feet)
0	30
10,000	60
20,000	70
30,000	80
40,000	120

NOTE: Each altimeter and its scale error corrector are calibrated as a set and have the same serial number. If either unit is removed from the airplane, both units must be replaced with a matched set.

Machmeter

Two Machmeters are provided, one for the pilot and one for the copilot. Each Machmeter gives a continuous indication of airspeed in terms of Mach number.

A differential pressure diaphragm and a static pressure diaphragm, interconnected by a mechanical linkage, sense the two pressures, solve the ratio between them mechanically and indicate the Mach number by a radial pointer moving over a dial calibrated in percentage of Mach number. A synchro transmitter mounted on the pointer shaft transmits a Mach signal to the static system error corrector of the KIFIS. The mechanism is compensated for variations in temperature ranging from -55 degrees C to +70 degrees C (-67 degrees F to 158 degrees F).

True Airspeed Indicator

A true airspeed indicator is located on the pilot's instrument panel. The true airspeed is indicated by a digital display through a window in the face of the instrument.

True airspeed is a function of Mach number and static or true outside air temperature. An electromechanical computer in the control chassis assembly combines these functions to actuate the motor which operates the counters in the instrument.

WARNING: THE TRUE AIRSPEED INDICATOR IS NOT TO BE USED BELOW 200 KNOTS.

Static Air Temperature Indicator

A static air temperature indicator is located in the center instrument panel. The true outside air temperature is indicated by a radial pointer moving over a scale with a range of -100 degrees C to +50 degrees C (-147 degrees F to 122 degrees F).

The outside air temperature is sensed by a temperature bulb located in the slipstream of the airplane. In flight, the temperature indication would be higher than the true static temperature due to the compression of the air around the bulb. An electromechanical computer in the instrument applies a Mach correction signal to the signal from the temperature bulb to provide an accurate indication of the true air temperature.

KIFIS Test Switches

Test switches, marked KIFIS, are located on both the pilot's and copilot's instrument panels. These switches have TEST and NORMAL positions. In the TEST position, certain preset values for altitude, a true airspeed, and static air temperature will be presented on the respective instruments if they are operating properly. Operating the pilot's KIFIS test switch in flight may have a slight effect on the Mach Trim system.

FLIGHT DIRECTOR SYSTEM (BENDIX)

The flight director system provides the pilots with an instrument display showing heading, attitude, and deviation from selected VOR/LOC courses and ILS glide paths. The system also displays computed pitch and roll commands to assist the pilots in following the selected course. Input signals are received from the gyro compass and the BHF navigation and glide slope systems. Basically, the flight director system consists of two course deviation indicators, two horizon director indicators, two flight instrument amplifiers, a flight steering computer, and two vertical gyros (see Figure 18-3).

A course deviation indicator and a horizon director indicator are installed in the pilot's and copilot's flight instrument panels. Each course deviation indicator displays the compass heading and displacement from the glide slope path or the selected radial of a VOR or LOC station. The horizon director indicator indicates the pitch and roll attitude of the airplane, and displays computed command indications by a command bar.

The flight instrument amplifiers, one for each set of instruments, provide attitude and compass signal amplification to operate their respective course deviation and horizon director indicators. The flight steering computer coordinates compass heading, vertical gyro, and radio signals from the VHF navigation radio receivers to provide pitch and roll steering signals for the command bars of the two horizon director indicators.

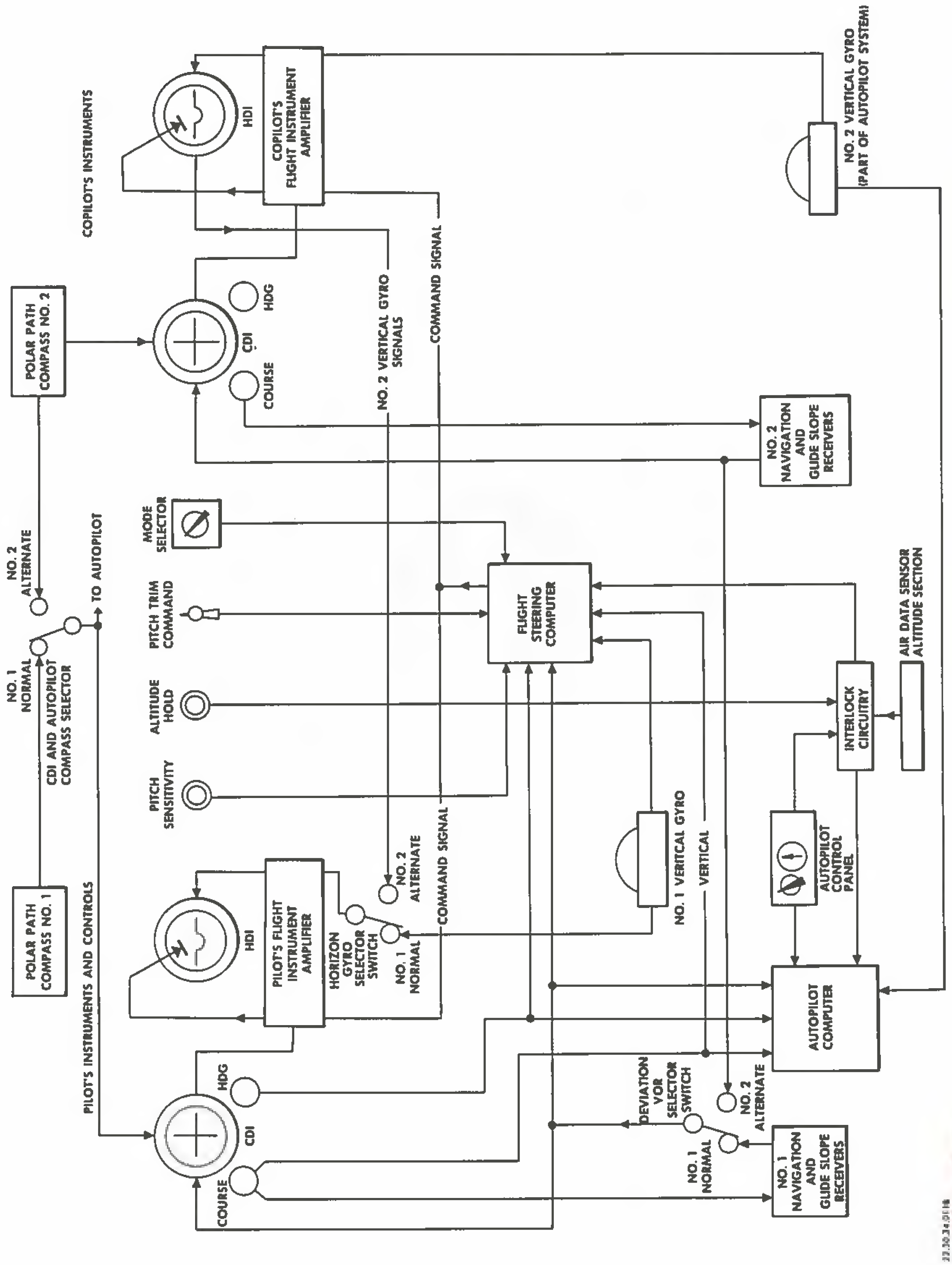
The No. 1 vertical gyro normally provides input attitude signals for the pilot's horizon director indicator; the No. 2 vertical gyro, which is part of the autopilot system, provides input signals to the copilot's horizon director indicator. A horizon gyro switch allows the pilot to select alternate attitude signals from the No. 2 vertical gyro.

Course Deviation Indicator (CDI)

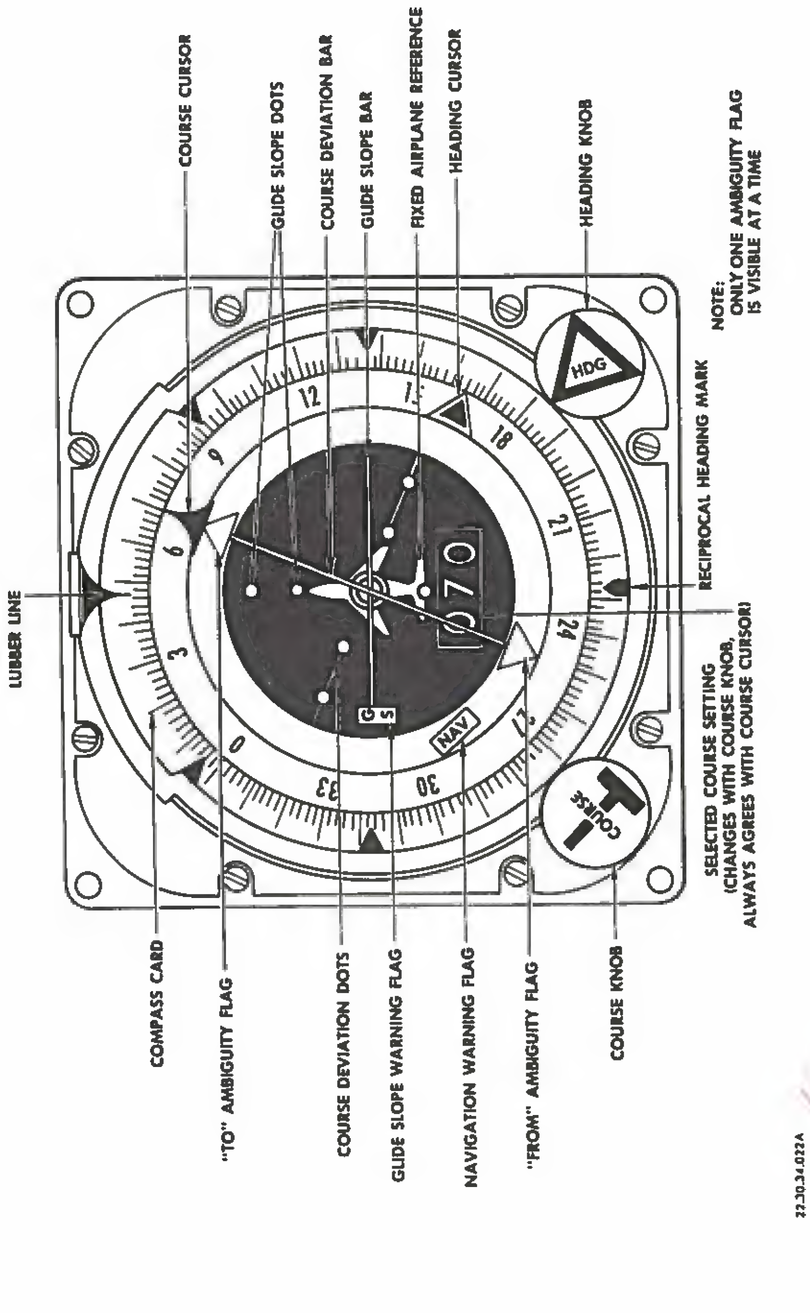
The case of the CDI carries a lubber line and reference marks 45 degrees and 90 degrees to the right and left of center and a reciprocal heading index at the bottom of the case (see Figure 18-4). A fixed center dial carries an image of an airplane, a window displaying the selected course, and vertical and horizontal reference dots. The moveable parts of the display include a compass card, heading and course cursors, and a deviation bar with ambiguity flags. heading and course selection knobs and malfunction flags complete the display.

The compass card is mounted on a rotatable carriage connected to the gyro compass components of the polar path compass system. The card is graduated in 2-degree increments clockwise from 0 degree to 360 degrees.

The magnetic heading is indicated by the position of the card with reference to the lubber line. The heading cursor can be positioned any place around the card by the HDG knob. When operating in the HDG mode, the autopilot will maintain the heading indicated by the heading cursor.



Flight Director System Diagram
Figure 18-3



Course Deviation Indicator
Figure 18-4

NOTE: Whenever heading information is required by the pilots, the two Bendix systems (two RMI and two CDI indicators) should be frequently checked against each other and against the standby magnetic compass to detect obvious errors in indication.

The selected radial of an OMNI range station or the bearing of the localizer beam of an ILS is set in the instrument by the COURSE knob. This knob positions the T-shaped cursor and deviation bar with reference to the compass card and rotates the counter in the window in the center dial. The numbers on the counter always agree with the bearing indications of the course cursor and deviation bar.

As the course knob is turned, the T-shaped cursor will move over the graduations on the compass card and indicate the direction displayed in the course counter window. Simultaneously, the course deviation bar will rotate over the center dial in line with the course cursor. When the VHF navigation radio is tuned to the selected VOR station, the deviation bar will move to the right or left indicating the position of the selected radial with reference to the airplane. An ambiguity flag is attached to each end of the deviation bar but only the one pointing toward the station will be visible.

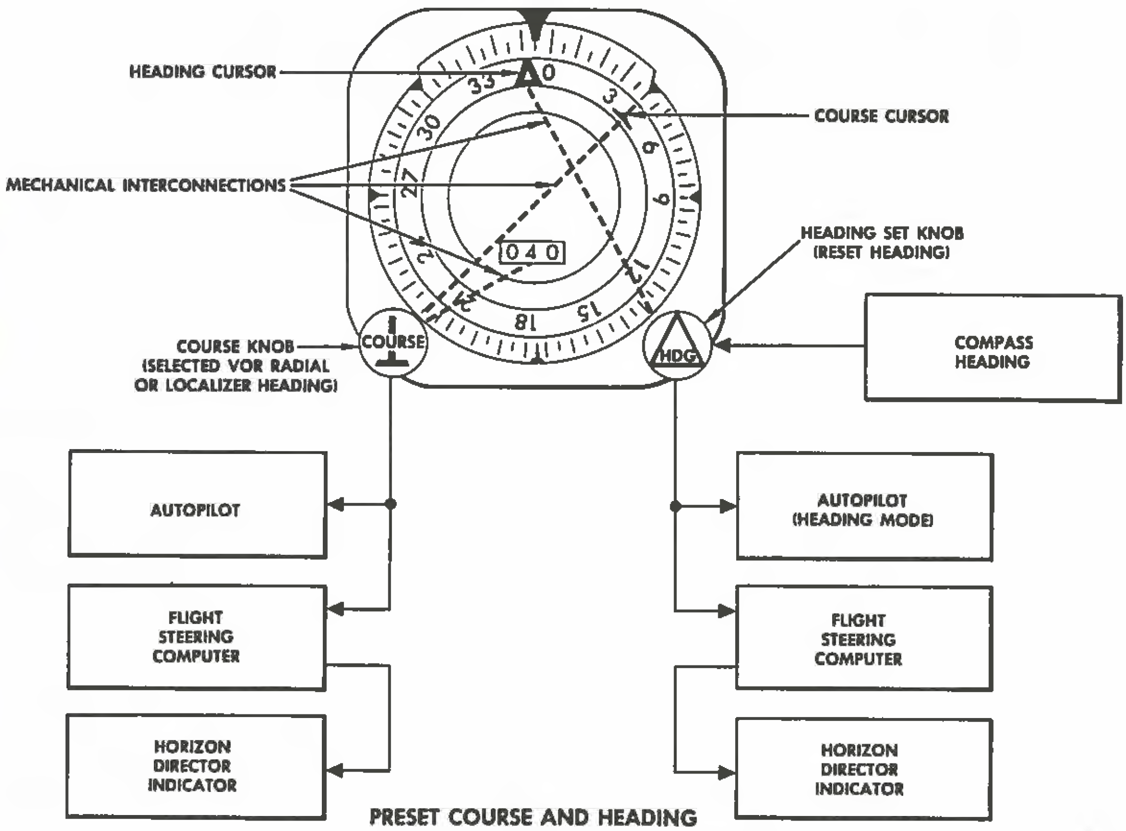
After the course cursor and the deviation bar have been set, they will maintain the same directional indication with relation to the compass card. Thus, in operation, the heading cursor, the course cursor, and the deviation bar all maintain a fixed directional position with respect to the compass card, regardless of changes in the heading of the airplane.

When making an ILS approach, the course counter and cursor are set to correspond to the bearing of the localizer beam and the deviation bar will move right or left of center to indicate the location of the localizer beam.

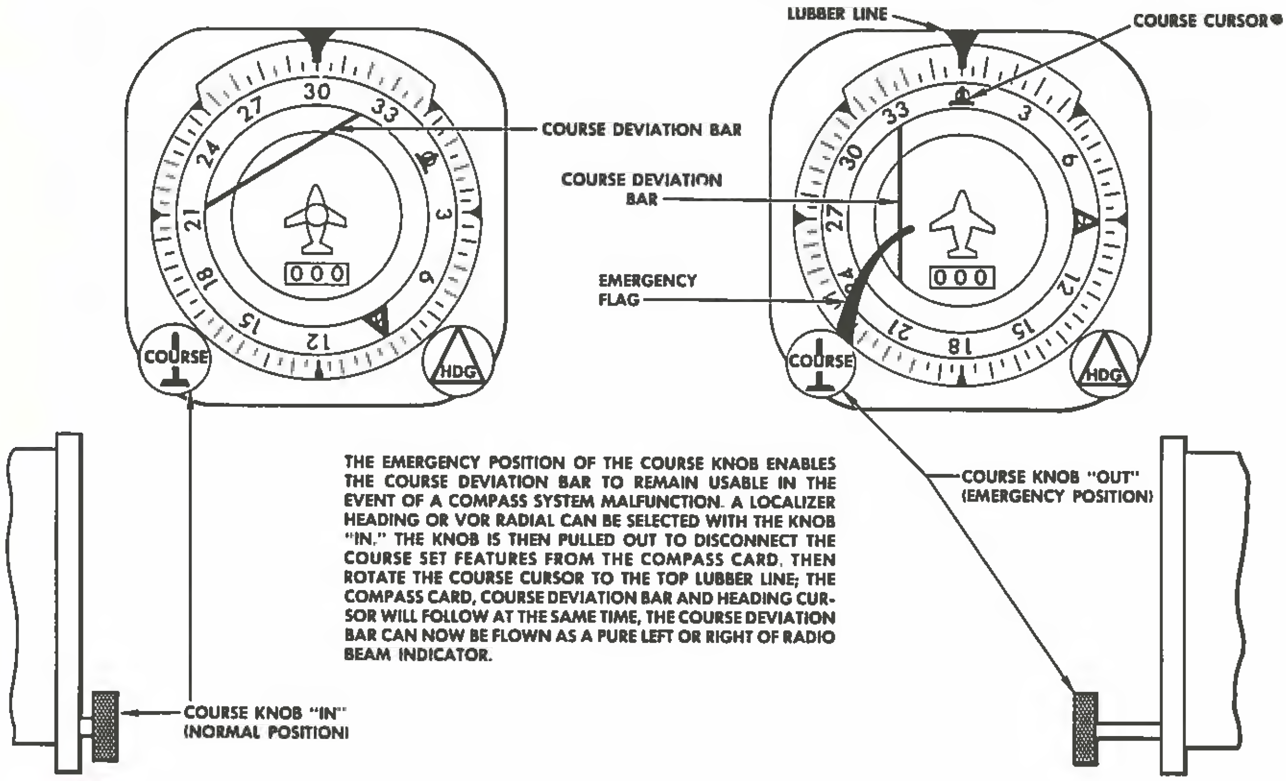
The glide slope bar is used only when making ILS approaches. When the VHF navigation receiver is tuned to an ILS station, the glide slope receiver is turned on automatically and the glide slope bar will move up or down from center to indicate the location of the glide slope path in relation to the position of the airplane. When the glide slope receiver is not in use, the bar is locked across the center of the dial. If the glide slope signal is lost, a GS flag will appear on the left side of the center dial. Both ambiguity flags are hidden when the instrument is being used for ILS operation.

The course knob of the course deviation indicator has two positions: the normal position which is used to select the desired VOR radial or the bearing of a localizer beam, and an emergency position in which the knob is pulled out from the case front. If the compass part of the instrument should malfunction, the course deviation components of the system can still be used (see Figure 18-5).

When the course knob is pulled out to the emergency position, the course selection features are disengaged from the knob, and the compass dial servo is de-energized. In this position, the knob drives the compass card and the course



PRESET COURSE AND HEADING



THE EMERGENCY POSITION OF THE COURSE KNOB ENABLES THE COURSE DEVIATION BAR TO REMAIN USABLE IN THE EVENT OF A COMPASS SYSTEM MALFUNCTION. A LOCALIZER HEADING OR VOR RADIAL CAN BE SELECTED WITH THE KNOB "IN." THE KNOB IS THEN PULLED OUT TO DISCONNECT THE COURSE SET FEATURES FROM THE COMPASS CARD, THEN ROTATE THE COURSE CURSOR TO THE TOP LUBBER LINE; THE COMPASS CARD, COURSE DEVIATION BAR AND HEADING CURSOR WILL FOLLOW AT THE SAME TIME, THE COURSE DEVIATION BAR CAN NOW BE FLOWN AS A PURE LEFT OR RIGHT OF RADIO BEAM INDICATOR.

COURSE KNOB—NORMAL AND EMERGENCY POSITIONS

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Course Deviation
Indicator Interconnections
Figure 18-5

cursor and deviation bar with their relative positions unchanged. The course cursor can then be aligned with the lubber line and the course deviation bar will continue to indicate the position of the airplane right or left of the localizer beam. In this operating mode, a red flag will appear between the inner dial and the compass card.

Horizon Director Indicator (HDI)

The HDI is conventional gyro horizon with a command bar added (see Figure 18-6). The command bar, operating in response to computer signals, furnishes the pilot with a simple method of controlling the attitude of the airplane to follow a predetermined flight program. The face of the instrument displays a gyro stabilized attitude sphere and roll index, a fixed horizontal airplane symbol, a command bar, a pitch trim knob, and malfunction flags.

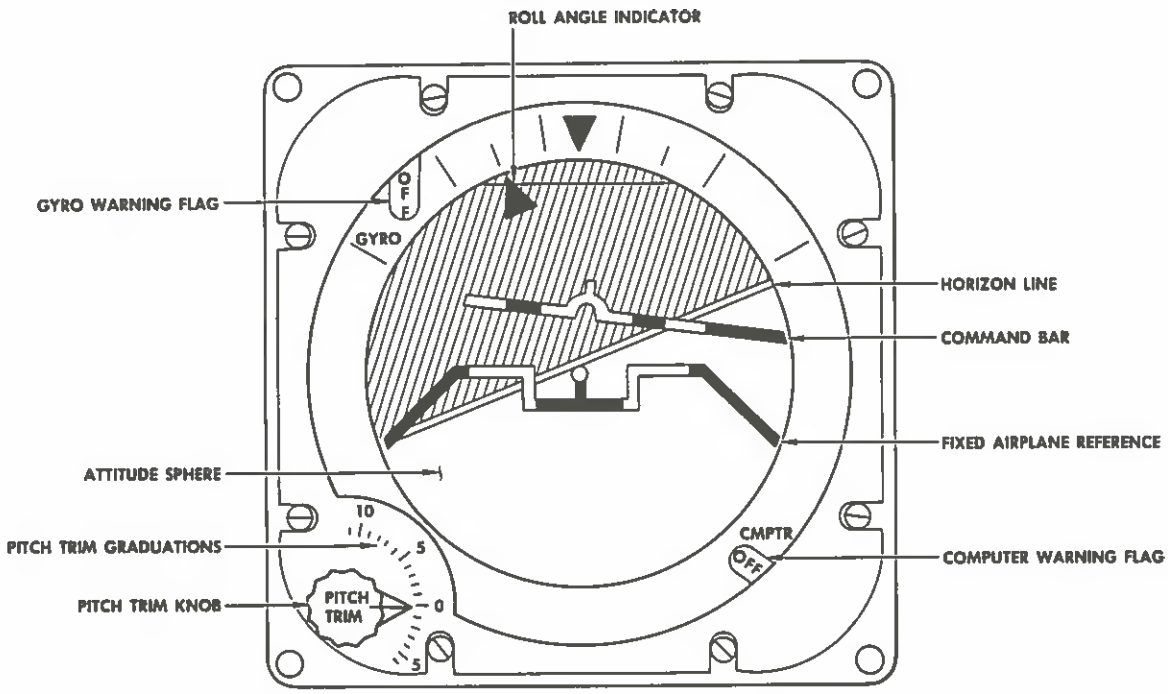
The attitude sphere is divided into blue and black segments by a horizon line and is free to rotate 360 degrees in roll and 80 degrees in pitch. The attitude of the airplane is indicated by the angle and vertical displacement of the horizon bar with reference to the fixed airplane symbol. The angle of bank is indicated by the position of the roll index in relation to the graduations along the top of the instrument case.

The attitude sphere can be raised or lowered by the PITCH TRIM knob at the lower left corner of the instrument to trim the horizon line for nose up or nose down flight attitudes. The pitch knob can be set to provide any pitch trim angle between 11 degrees nose up and 6 degrees nose down. The GYRO OFF flag becomes visible when the gyro malfunctions or is turned off.

The command bar comes into view when the HDI is used as a flight command instrument. The bar responds to roll, pitch, heading, altitude, radio navigation, and glide slope information and moves independently of the airplane symbol and attitude sphere. The command bar functions are selected through an altitude hold switch and a flight director mode switch which channels heading, VOR/LOC, or glide slope information to the instrument. When the selector switch is OFF, the flight steering computer is disconnected and the bar is automatically recessed out of view. The Flight Director system will also disconnect when the control wheel autopilot disengage switches are operated. The command bar can be trimmed by a trim rheostat located on the pilot's instrument panel.

In response to signals from the steering computer, the command bar will assume an attitude indicating the change in airplane attitude required to follow the selected flight program. Thus, the pilot can follow the selected VOR radial, the ILS localizer and glide slope signals, or the selected heading, or maintain a constant pressure altitude by flying the airplane to make the airplane symbol in the HDI conform to the attitude changes indicated by the command bar.

A NORM-HI sensitivity switch on the pilot's flight instrument panel enables the pilot to vary the sensitivity of the command bar display. Placing the switch in the HI position increases the sensitivity of the command bar and magnifies small deviations from the selected course or glide slope path. The NORM position provides normal pitch and roll correction commands.



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Horizon Director Indicator
Figure 18-6

If the flight steering computer should malfunction, the CMPTR OFF flag will come into view (see Figure 18-7).

Flight Steering Computer

The flight steering computer coordinates compass heading, vertical gyro, and radio signals to provide roll and pitch steering signals for the command bar of the horizon director indicators. The computer contains an ac and dc power supply and various plug-in type card assemblies for computing the command bar signals. The steering computer is installed in a shock-mounted rack in the autopilot compartment.

Flight Instrument Amplifiers

Two flight instrument amplifiers, one for each set of indicators, provide attitude and compass signal amplification to operate their respective course deviation and horizon director indicators. Basic power is supplied by 115-volt ac airplane power and integral power supplies provide the necessary ac and dc voltages for system operation. The flight instrument amplifiers are installed in the shock-mounted autopilot rack.

Vertical Gyro

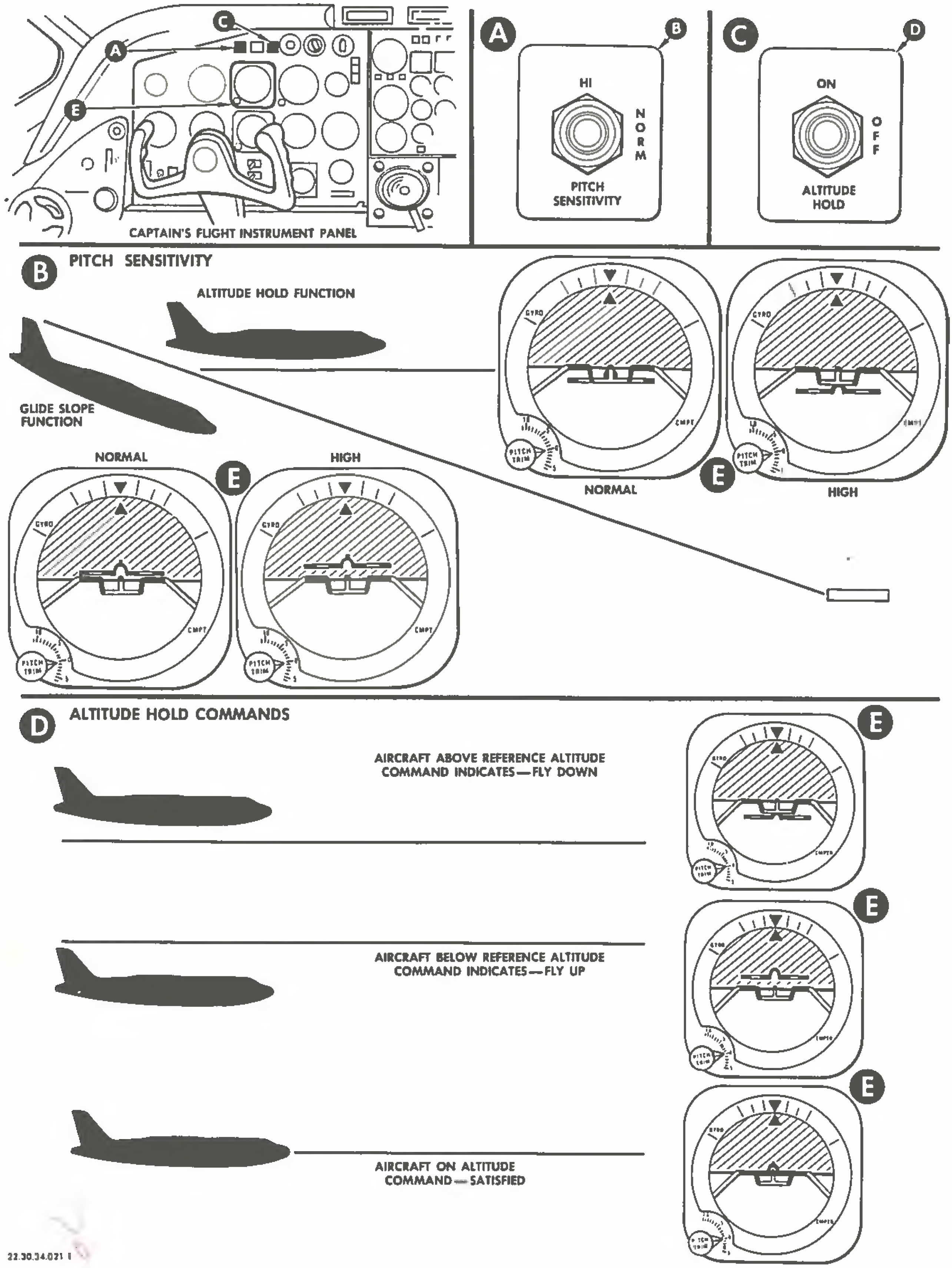
The output signals from two vertical gyros are fed into the flight director system. The No. 1 gyro operates the pilot's flight instrument amplifier; the No. 2 gyro operates the copilot's flight instrument amplifier and the autopilot. The gyro rotors are driven electrically and are free to rotate 360 degrees in the roll plane and 85 degrees in the pitch plane with controlled tumbling beyond the pitch stops. An internal electric circuit provides for initial gyro erection and eliminates the need for a manual caging device. The gyro is ready for use within one minute after being turned on.

A horizon gyro selector switch marked No. 1 NORM and No. 2 is located on the pilot's instrument panel. With the switch in the NORM position, the pilot's HDI receives attitude signals from the No. 1 gyro. In the No. 2 position, the pilot's HDI is connected to the copilot's HDI which receives its attitude signals from the No. 2 gyro in the autopilot.

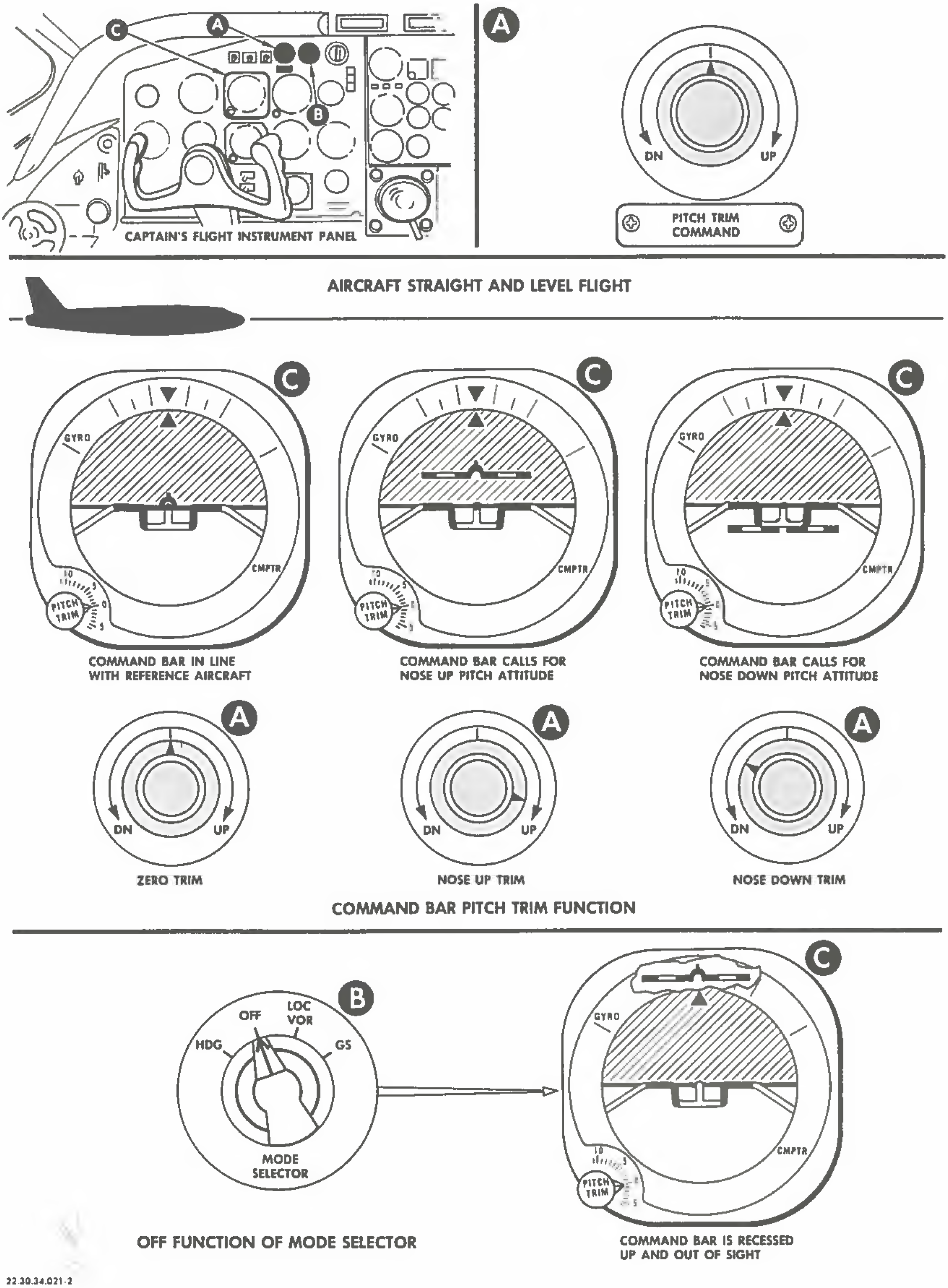
SEARCH-WEATHER RADAR SYSTEM

The search-weather radar system functions primarily to detect severe storm areas. It also serves as an anti-collision warning device through detection of objects in the flight path of the airplane. It may also serve as a navigational aid by providing terrain mapping. The search-weather radar system is comprised of an antenna-reflector, receiver-transmitter, accessory unit, control panel, and an indicator scope (see Figure 18-8).

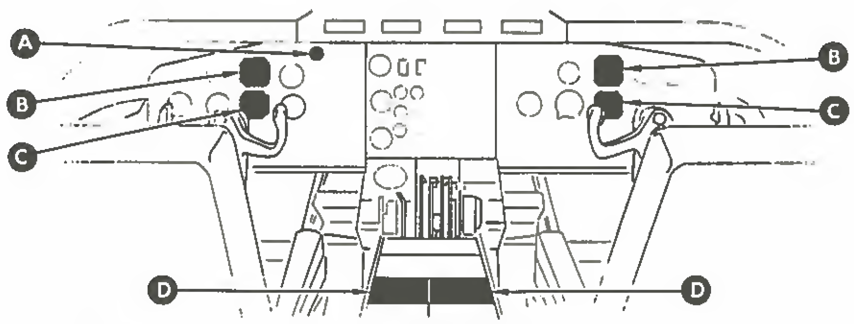
Short high-power bursts of electrical energy are fed to the antenna from the transmitter and are propagated outward in straight lines with reference to the reflector position. When these energy impulses encounter an object of sufficient

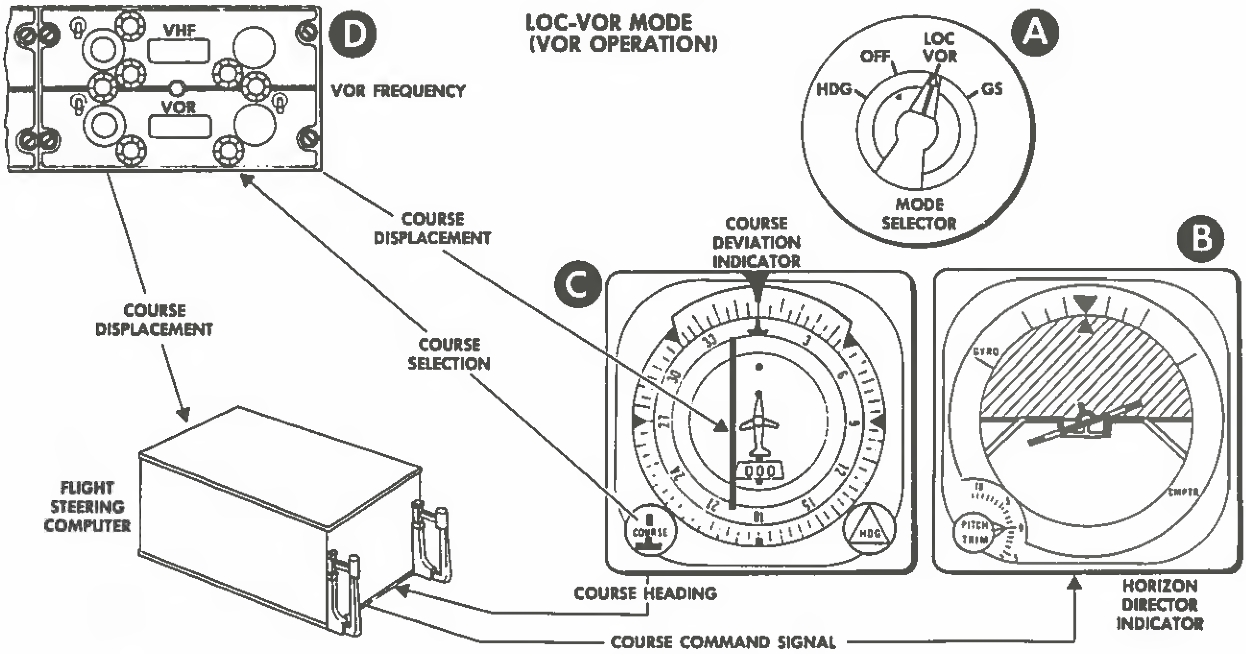


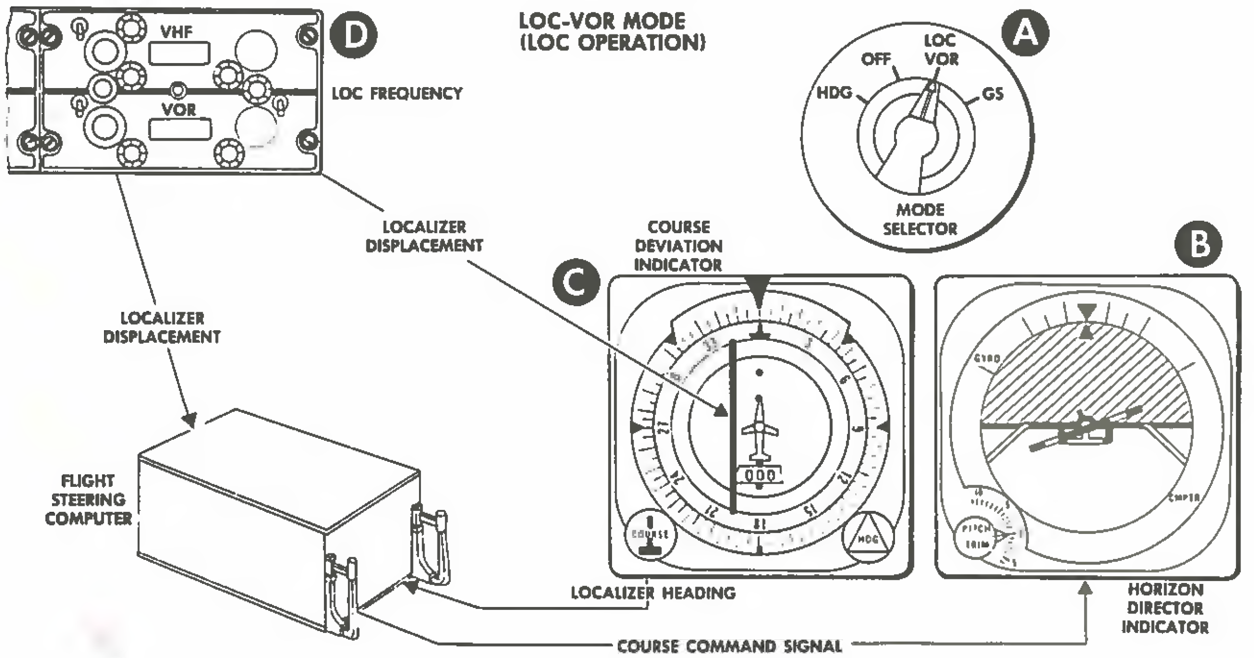
Flight Director System Operation
Figure 18-7 (Sheet 1 of 4)



Flight Director System Operation
Figure 18-7 (Sheet 2 of 4)







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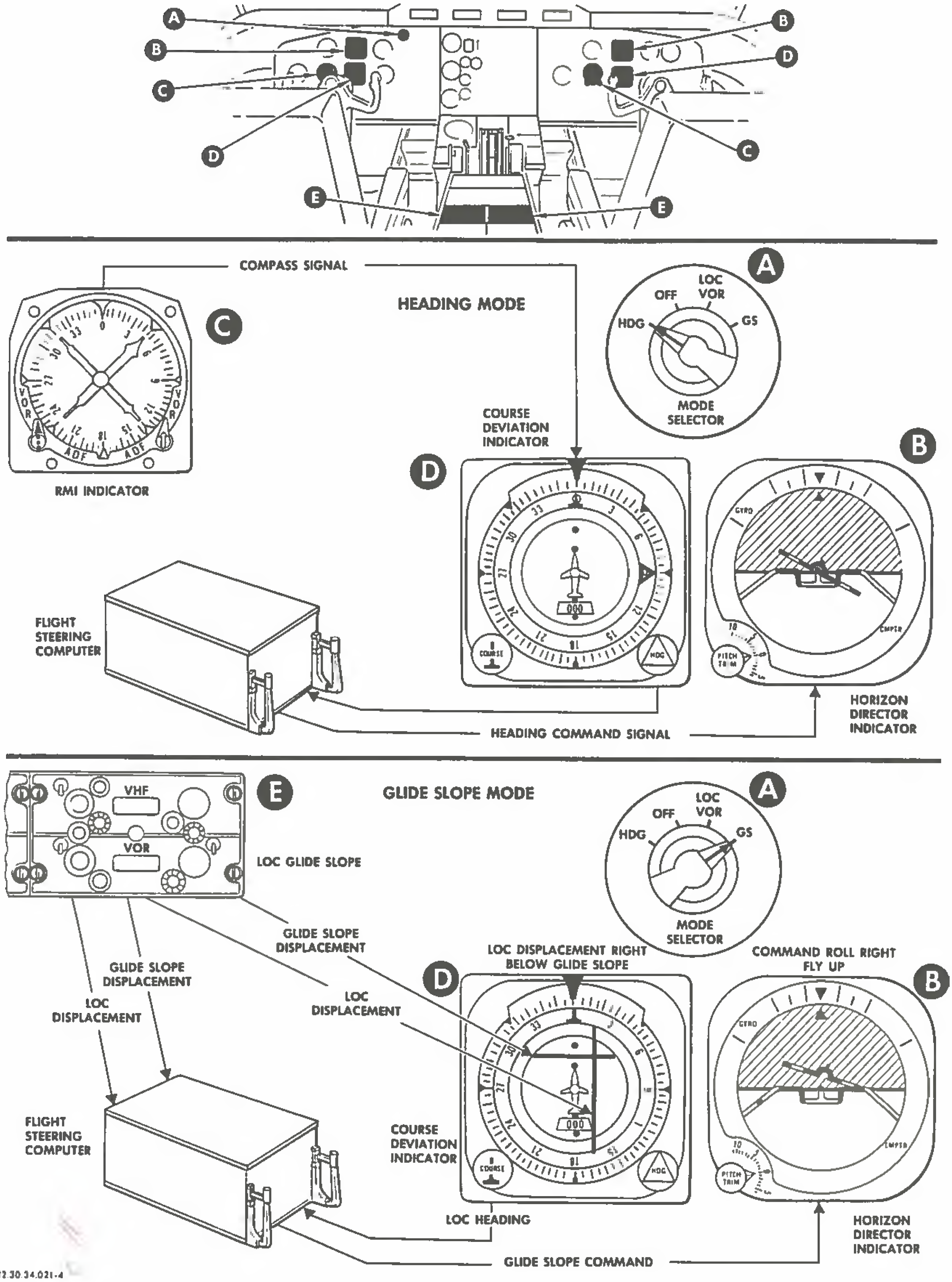
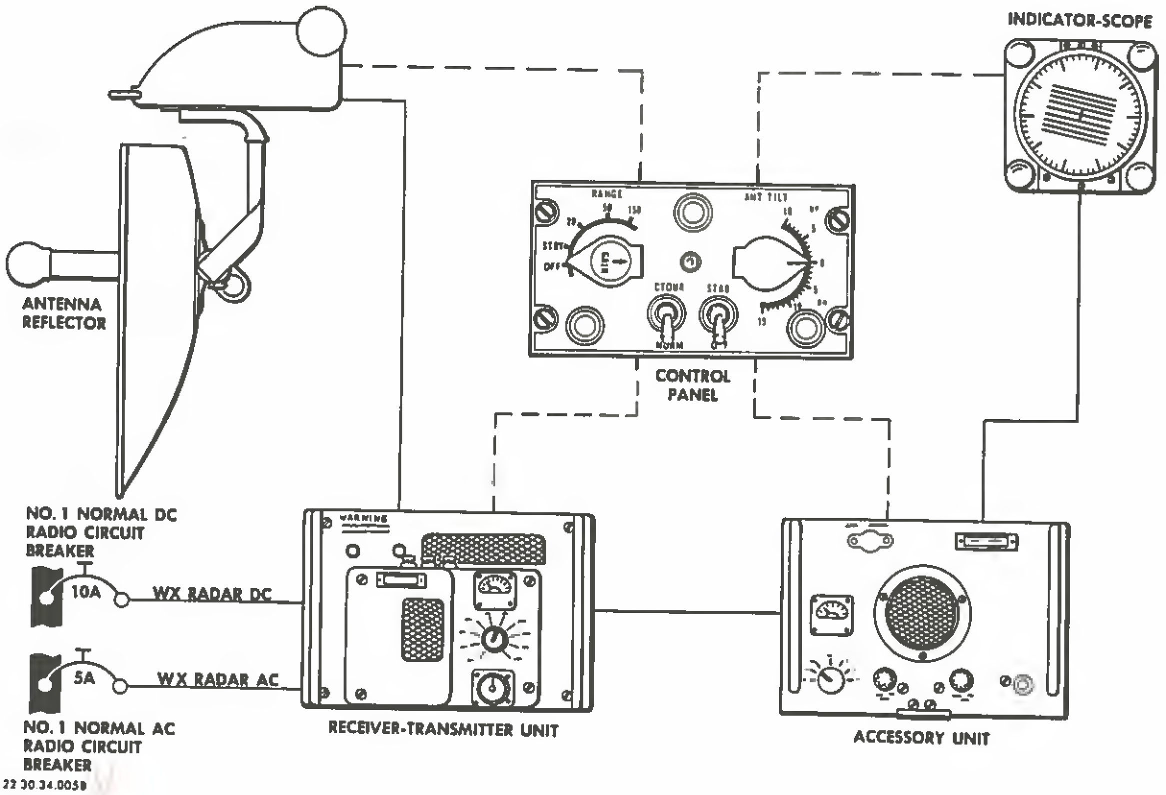
Flight Director System Operation
Figure 18-7 (Sheet 3 of 4)



Flight Director System Operation
Figure 18-7 (Sheet 4 of 4)



Search-Weather Radar Diagram
Figure 18-8

density, the pulses are reflected back to the antenna. These reflected pulses are amplified by the receiver and accessory units and are measured for display on the indicator-scope. The time interval between the transmitted and received pulses provides the basis for distance determination.

Simultaneously, the direction in which the reflector is pointed causes an azimuth indication to appear on the indicator scope. Since the pulses are transmitted in the horizontal plane as the indicator sweep is rotated about its origin, the distance of the target and its direction away from the airplane are displayed on the indicator scope.

Receiver-Transmitter

The transmitter supplies 75 KW minimum peak power to the antenna at a frequency of 5400 megacycles. In addition, the transmitter supplies the accessory unit with the automatic frequency-controlled received signal information. A running time meter, a test meter, and a switch with eight metering positions are installed on the front panel. The range can be set by a RANGE switch on the control panel at 20, 50, or 150 miles.

Accessory Unit

The accessory unit contains the synchronizing, video, iso-contour, I-F, and stabilization circuitry for the radar system. In addition, the accessory unit provides filament and plate power for one or both of the indicators as well as the preamplifier and automatic frequency control unit in the receive-transmitter. Two fuses, a test meter and an eight position meter selector switch are provided on the control panel.

Indicator Unit

The indicator unit houses the five-inch cathode-ray tube on which target information is displayed. The indicator also contains the final video amplifier, yoke-rotation mechanism, range marker lamps, and the high voltage power supply for the cathode-ray tube. The four adjustments on the indicator front assembly provide for RANGE MARKS, LIGHT, SWEEP, CURSOR, and INTENSITY GAIN.

Antenna-Reflector Assembly

The antenna-reflector is a lightweight transmit-receive unit that is line-of-sight stabilized against airplane pitch and roll. The antenna assembly consists of a base assembly housing, a synchro motor and transmission, a reflector frame, and a reflector feed dust cover on which the reflector assembly is mounted. The reflector assembly consists of the reflector, feed, matching plates, and mounting hardware.

TERRAIN WARNING SYSTEM

The terrain warning system utilizes a radar beam transmitter-receiver and an antenna combination to provide visual and aural warnings of terrain obstacles.

Transmitter-Receiver Unit

The transmitter-receiver consists of a transmitter whose RF out-put radiates from the antenna and a receiver which picks up reflections on the same antenna from objects on the ground. The receiver converts the reflected signals into a form suitable to operate the indicator lights or the aural warning device.

Antenna

The antenna is mounted on the bottom of the forward fuselage and serves as a receiving and transmitting unit. The body of the airplane serves as an effective shield against upward radiation from the antenna.

Distance Selector Switch and Indicator Lights

The distance selector switch and indicator lights are mounted at the left side of the pilot's flight instrument panel. The selector switch is a 3-position switch labeled 2000, 1000 and 500 feet. Three associated indicator lights are labeled 2000, 1000 and 500 feet. With the switch set at any of the three positions, the corresponding warning light will come on when the airplane comes within the selected distance from the ground. A warning bell will sound when one of the warning lights illuminates.

Test Switch

The warning lights will illuminate and the warning horn will sound when the test switch is placed in the ON position and the system is operating properly.

AUTOMATIC DIRECTION FINDING SYSTEM (ADF)

The ADF system, sometimes called the Dual Radio Compass System, is used for automatic direction finding, manual direction finding, and radio range reception. Directional guidance is presented visually on an indicator on the flight instrument panel, aurally on the flight interphone system, or both can be presented simultaneously. The dual ADF system is designated ADF-1 and ADF-2. Both systems are identical.

Each ADF system incorporates a receiver located in the radio rack, a control unit in the flight compartment, a motor-driven directional loop antenna mounted on the bottom of the fuselage beneath the forward baggage compartment, an indicator located on the flight instrument panel, a non-directional antenna located on top of the fuselage, and an antenna coupler.

ADF Receiver

The compass receiver is an airborne navigation aid designed primarily for use as an aircraft automatic direction finder. It furnishes both aural and visual signals to the pilot. The compass receiver is a multi-channel super-heterodyne covering a frequency range of 90 to 1800 kilocycles in four bands.

ADF Remote-Control Unit

A remote-control unit controls all the functions of the ADF system. It controls band selection and tuning, and provides a digital indication of the selected frequency. A selector switch is provided for selection of the ADF, ANT, and LOOP modes of operation.

A LOOP switch is used to rotate the loop antenna, either right or left, when the system is operating in the LOOP mode. A gain or volume control knob is provided for adjusting the audio level. Two toggle switches are located on the upper face of the control unit; one switch turns the beat frequency oscillator on and off, the other switch is used for selecting broad or sharp tuning. A tuning meter on the unit provides a visual tuning indication for maximum reception.

ADF Antennas

Two antennas are provided for each system. A non-directional antenna provides aural reception of broadcast and range stations and directional information from radio range stations. The loop antenna provides aural-null directional indications and directional signals to position the pointers in the indicator. The unit provides reception of either modulated or keyed CW by operating a switch on the control head. In the ADF mode compass operation is entirely automatic, and it is necessary only to tune the receiver to the frequency of the selected radio station in order for the bearing of that station to be indicated directly on the bearing indicator.

Loop Antenna Rotation

The loop antennas are rotated by reversible electric motors. When one of the selector switches is placed in the LOOP mode, the respective loop can be rotated to the right or left by the LOOP switch. With the selector switch in the ADF position, ambiguity is automatically resolved and the loop is rotated to the proper null position.

ADF Indicators

Each ADF indicator or radio magnetic indicator (RMI), is a panel-mounted instrument containing a remotely-controlled rotating azimuth card, a single and a double-barred synchro-operated pointer, and two selector switches. The card is positioned by signals from the polar path compass system and normally indicates the magnetic heading of the airplane. When the polar path compass is being operated in the directional gyro (DG) mode, the card will indicate the direction in which the gyro is oriented.

The single pointer is positioned by the loop antenna of the ADF-1, or pilot's system, and indicates the bearing of the radio station to which the ADF-1 receiver is tuned. The double-barred pointer is positioned by the loop of the ADF-2 system and indicates the bearing of the station to which the ADF-2 receiver is tuned.

Two selector switches, marked ADF-VOR, are located at the bottom of each instrument case. The switch on the left has a line representing the single needle and is positioned to connect the single needle to the ADF-1 or the VHF-1 navigation receiver. The switch on the right has a double line and connects the double-barred needle with the ADF-2 or VHF-2 navigation receiver. Thus, cross bearings can be taken by each indicator on two low frequency stations, on two VHF stations, or on a low frequency and a VHF station.

MARKER BEACON SYSTEM

The marker beacon system provides the pilots with visual and aural indication of signals received from fan markers located along the airways, from Z markers located at the range stations, and from the middle and outer ILS markers. The fan markers emit a 3000-cycles-per-second tone signal which is keyed with Morse code dots and dashes for identification. The Z markers transmit a 3000-cycles-per-second tone signal. The outer ILS marker transmits a 400-cycles-per-second tone signal keyed in Morse code dashes. The middle marker transmits a 1300-cycles-per-second tone signal keyed in alternating dots and dashes. All of the markers transmit on a frequency of 75 mc.

Indicator Unit

The marker beacon system consists of an antenna located on the bottom of the fuselage, a receiver mounted on the radio rack, two sets of indicator lights, and a sensitivity control. Each set of indicator lights, one for the pilot and one for the copilot, is made up of three lights arranged vertically on each pilot's instrument panel. The top light is clear and is placarded AIRWAYS, the center light is blue and is placarded OUTER, and the bottom light is amber and is placarded MIDDLE.

Sensitivity Control

The sensitivity control is a 2-position toggle switch placarded HI and LO. When the airplane is flying at high altitudes, the switch is moved to the HI sensitivity position; when flying at low altitudes, the switch is moved to the LO sensitivity position. Thus, the sensitivity of the marker beacon receiver can be adjusted for signal strength variations due to high airplane altitude.

TRANSPONDER BEACON SYSTEM

The transponder beacon system assists the air traffic controller in identifying aircraft within his control area and in tracking aircraft through heavy ground clutter and precipitation.

The transponder beacon system utilizes an interrogator unit located at the air traffic control center and a transponder unit installed in the airplane. The ground interrogator is coupled to the airport surveillance radar and its antenna rotates in synchronism with the ASR antenna.

The three components of the transponder system are the "L" band stub mounted antenna on the bottom of the forward fuselage, the transponder unit in the radio rack, and the ATC control panel in the flight compartment.

DUAL DISTANCE MEASURING EQUIPMENT

Mount and wiring provisions are made for installing a single or dual set of distance measuring equipment.

POLAR PATH COMPASS SYSTEM

The polar path compass system utilizes a directional gyro and the output of a fluxgate compass. The directional gyro, corrected by manual setting for drift error, is available for use in grid navigation and rhumb-line flying. Alternately, the gyro may be slaved to the output of the pendulus fluxgate compass transmitter when flying with reference to a magnetic heading. Thus, the system provides two modes of operation; directional gyro (DG) with manual drift correction, and directional gyro slaved to the magnetic compass (SLA). The DG mode is used for navigation in the higher altitudes where the magnetic compass is unreliable while the SLA mode is preferred when operating in the middle and lower latitudes.

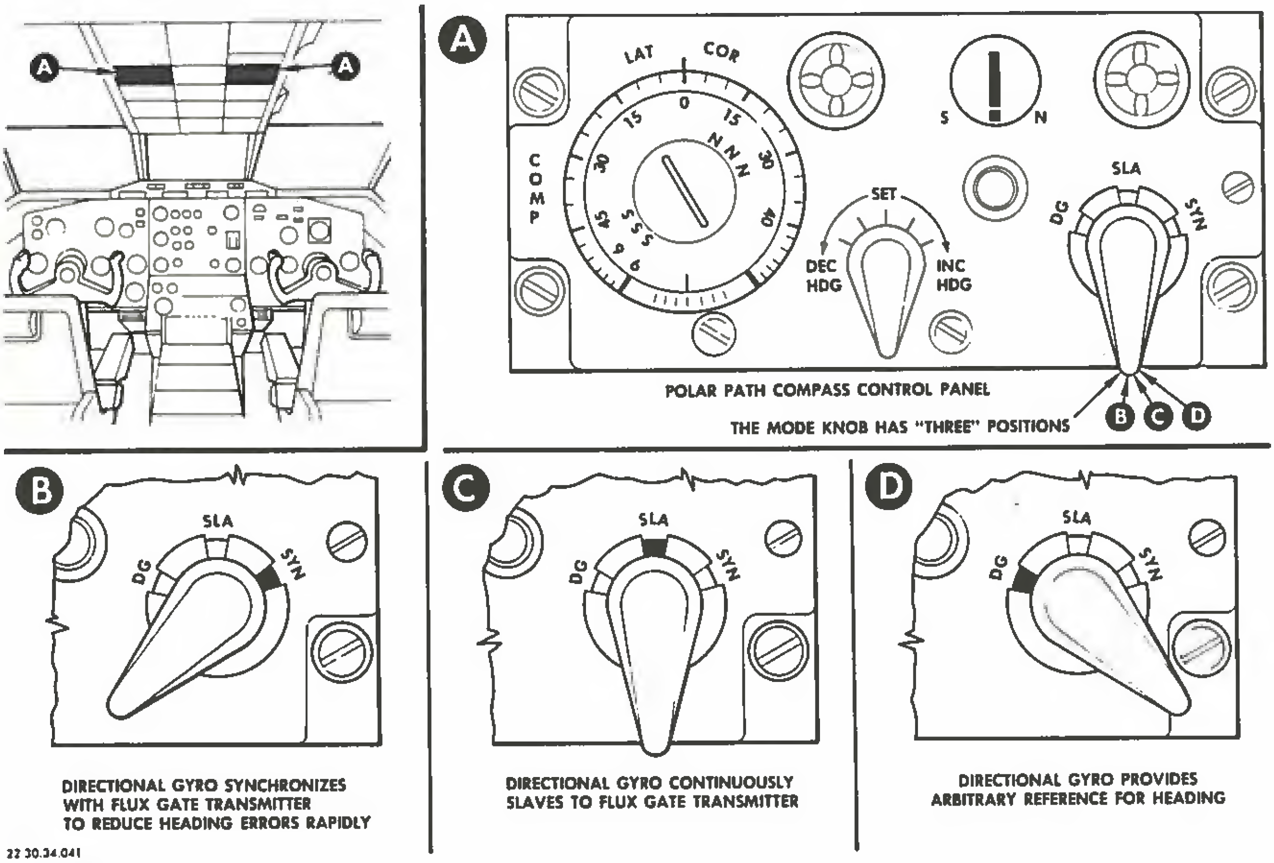
The polar path dual system supplies heading information to repeaters, radio magnetic (RMI) and course deviation (CDI) indicators, and heading control information to the automatic flight control system. Both pilots have separate polar path indicators and controls. Should the pilot's system fail, he may switch over to the copilot's system by actuating the transfer switch located on the pilot's flight instrument panel.

The polar path system components available for each pilot consist of the directional gyro and fluxgate transmitters, a compass coupler, and a panel-mounted compass controller.

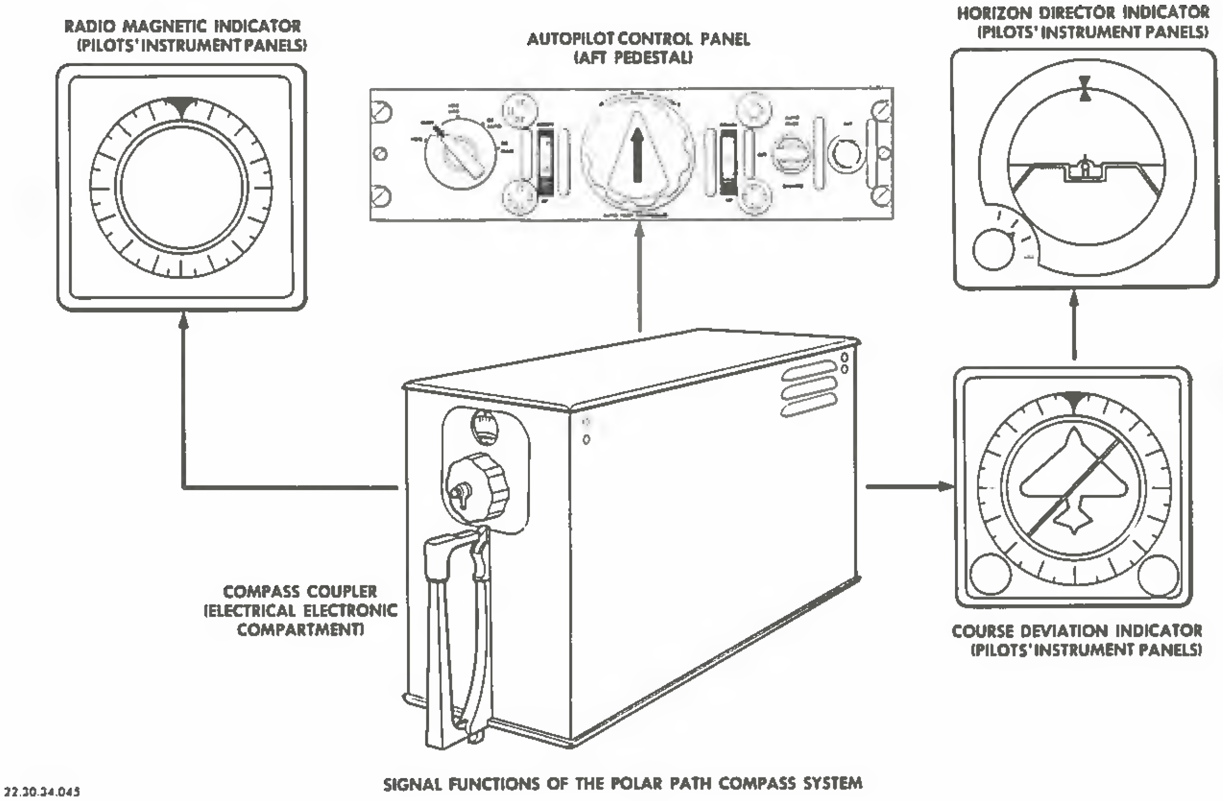
Compass Controller

Each pilot has a separate controller for his particular compass system. The controller panels are installed on the overhead switch panel of the airplane. Each controller consists of a mode selector switch, a course set switch, a synchronizer indicator and warning light, a rate of change or latitude correction knob, and three integral panel lights (see Figures 18-9, 18-10 and 18-11).

The mode selector is a 3-position switch with DG, SLA and SYN positions. The switch has a spring return from SYN to SLA position. In the DG position, the system has no magnetic correction applied to the directional gyro. In the SLA position, magnetic correction is applied to the directional gyro by the fluxgate transmitter. The SYN position is used to align the compass card with the output of the fluxgate transmitter.

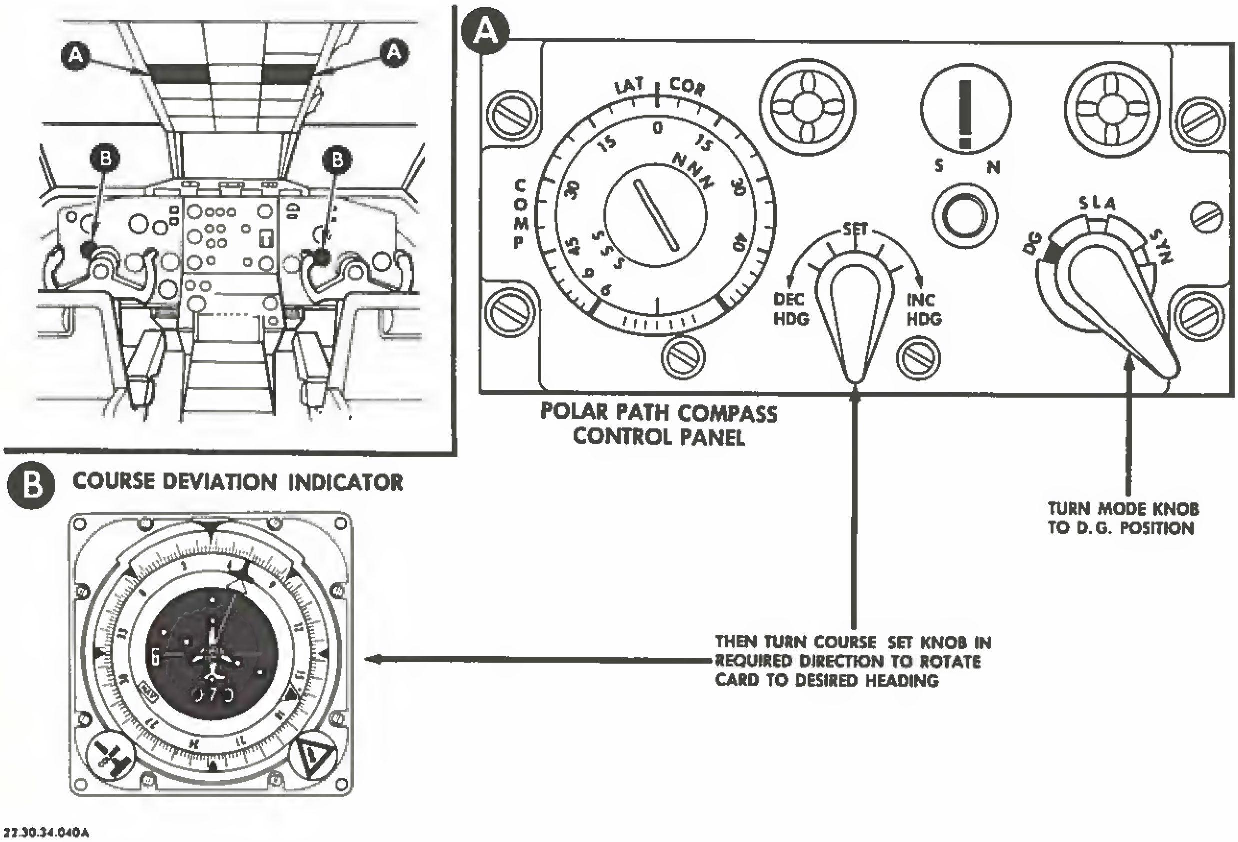


Polar Path Compass Control Panel
Figure 18-9



SIGNAL FUNCTIONS OF THE POLAR PATH COMPASS SYSTEM

Polar Path Compass
System-Signal Output Functions
Figure 18-10



Polar Path Compass System - Course Setting
Figure 18-11

The SET switch, located in the center of the controller panel, is used to set the heading reference in the indicator when the system is being operated in the DG mode. It may also be used to synchronize the compass card with the compass heading when the system is turned on or changed to SLA operation. The switch is spring-loaded to the center position and has two detents on each side of center. Turning the switch to the right will decrease the compass card indication, the compass card indication will increase when the switch is turned to the left. With the switch in the first detent, the card will rotate at 1/4 rpm; in the second detent, the card will rotate at 2 rpm.

The synchronizing indicator shows the degree of synchronization between the magnetic heading and the indicated heading when the mode selector switch is in the SLA position. When the magnetic heading and the indicated heading are synchronized, the needle of the indicator will be aligned with the reference mark at the base of the instrument. When the mode selector switch is in the DG position, the synchronizing indicator will indicate the relative amount and direction in which drift correction is being applied by the latitude corrector.

Directional Gyro Transmitter

The directional gyro is a low-drift horizontally-mounted gyro constructed for precision performance. The gyro is electrically driven at high speed by 3-phase 400-cycle ac power. An erection system is included in the unit to maintain the gyro spin axis in a horizontal plane. The complete assembly is enclosed in a sealed cover and the unit is filled with dry helium to prevent corrosion.

Any change is the heading of the airplane will cause the case of the instrument to rotate about the gyro rotor and cage which remain directionally fixed. Displacement of the gyro gimbal produced by the heading change results in the generation of autosyn synchro output voltages corresponding in direction and magnitude to the rotor displacement.

The synchronizer warning light will illuminate whenever the mode selector switch is in the SYN position or the set knob is displaced from center. If the light stays on after the switch is released, either the mode selector or the set knob did not return to its proper position.

The latitude selector knob, labeled LAT COR is used to set a circular dial numbered from 0 to 90 N and 0 to 90 S to indicate degrees of latitude. When operating in the DG mode, the directional gyro should develop an apparent drift due to rotation of the earth. When the latitude in which the airplane is operating is set in the latitude corrector, the directional gyro information will be corrected for the apparent drift. The synchronizing indicator should be checked to make sure that rate correction is being fed into the gyro.

Fluxgate Transmitter

The fluxgate transmitter is a remotely located device used to generate output voltages representing the magnetic heading of the airplane. It consists of a fluxgate element gimballed in pitch and roll, and a compass float containing two magnets. The magnets are pendulously mounted in pitch and roll but are free in azimuth. As the compass card is free to rotate in azimuth, the magnets remain aligned with the horizontal component of the earth's magnetic field. The fluxgate and compass elements maintain a horizontal attitude up to an airplane roll of 20 degrees.

The case is filled with compass fluid to provide damping for the pivoted elements. The fluxgate transmitter is powered by 400-cycle ac for excitation purposes.

Compass Coupler

The compass coupler is actually an amplifier-computer used to convert sensed heading data to electrical signals for airplane heading information and control. The unit consists of various plug-in card components which are used for amplification, frequency conversion, preamplification, and power supply.

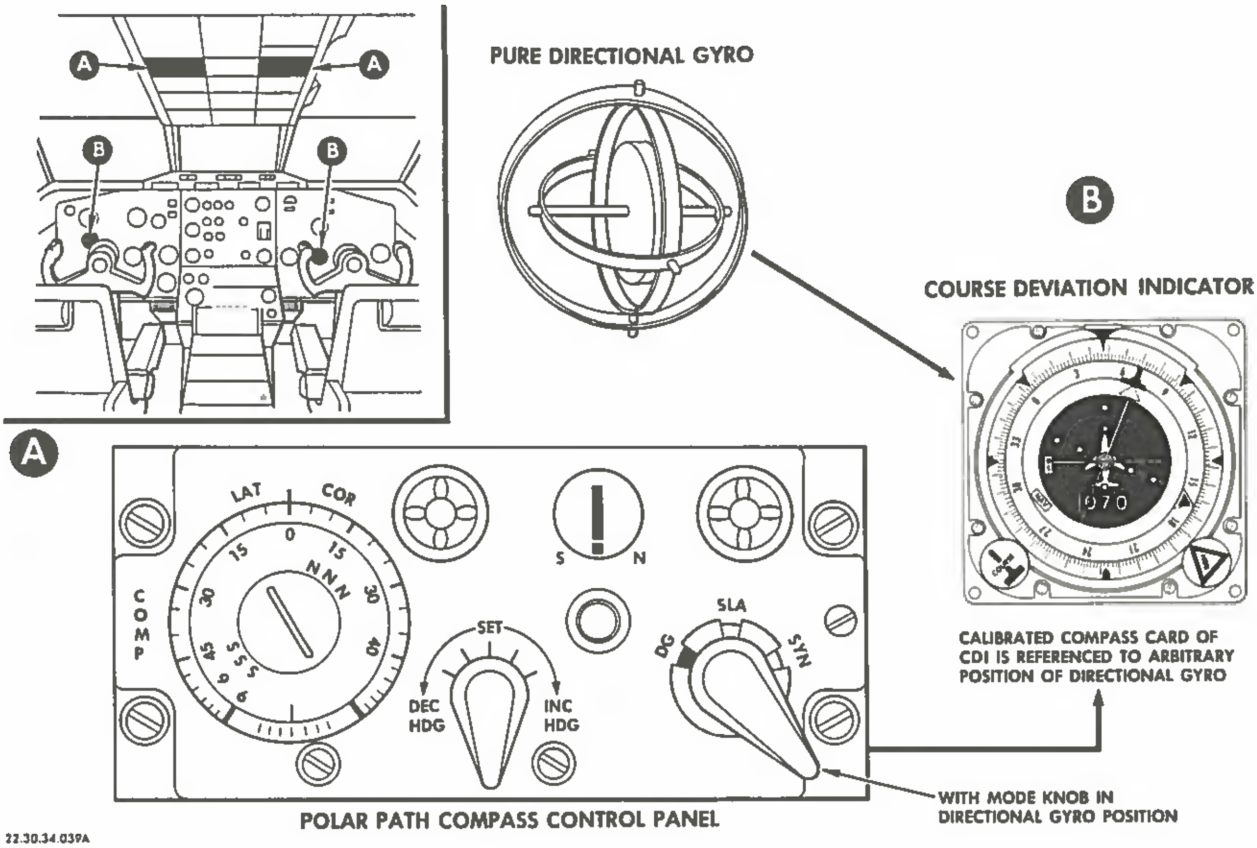
Compass compensating controls are accessible through an access door on the front of the panel case. These controls consist of a compensated and an uncompensated dial with 2-degree graduations and a vernier which permits one-half degree graduations to be read. Instructions for use of the vernier in dial readings are imprinted on the plate just below the access door.

Directional Gyro Operating Mode

The DG mode of operation uses only the directional gyro transmitter for directional reference (see Figure 18-12). As the directional gyro does not have any inherent directional properties, the initial heading must be set manually into the system. The appropriate heading is set in the radio magnetic indicator by the SET knob of the controller. Once the heading has been set in the indicator, the directional gyro transmitter develops autosyn output voltages which reposition the output data shaft in accordance with any changes in heading. Apparent gyro drift due to the earth's rotation is compensated for by the proper setting of the LAT COR knob on the controller.

Slaved Operating Mode

In the SLA mode, the directional gyro provides short term directional reference. This information is slowly slaved to the magnetic reference information from the fluxgate transmitter. This combination utilizes the most accurate characteristic of each directional reference.



Polar Path Compass System - Directional Gyro Mode
Figure 18-12

At initial engagement of the system, it is necessary to synchronize the directional gyro with the magnetic heading. This is accomplished by moving the selector switch to SYN for rapid synchronization of directional gyro information with that of the fluxgate transmitter, or by using the SET knob to set the gyro manually.

NOTE: Operating the SET knob, or moving the controller selector to the SYN position, automatically declutches the autopilot course autosyn synchro. However, the centering solenoid will not deenergize, thus enabling the course autosyn synchro to maintain any previously established trim signal.

MAGNETIC COMPASS

A standby magnetic compass is provided in the glare shield above the pilot's instrument panel. The compass is stowed in the glare shield when not in use. To use the instrument, it is lifted into the line of vision above the glare shield.

The standby compass indicates the heading of the airplane with reference to magnetic north. The compass card is graduated in 5-degree increments and numbered every 30 degrees. Damping fluid surrounds the compass card assembly to reduce oscillations and a small light illuminates the card. The light is turned on by a switch on the copilot's flight instrument panel.

The compass is compensated for deviation by adjusting a pair of permanent magnets located in the compass compensator below the compass bowl. The compass correction card showing the deviation remaining after the compass has been compensated is installed above the compass dial.

TURN AND SLIP INDICATOR

The turn and slip indicator is a combination of two instruments, a turn indicator and a laterally mounted inclinometer. The turn indicator consists of an electric-driven gyro and a pointer which indicates the direction and the rate at which the heading of the airplane is changing.

The inclinometer consists of a black ball free to move from side to side in a curved glass tube filled with a damping liquid. The position of the ball indicates the net result of the lateral forces acting on the airplane. When the ball is centered, the airplane is in perfect lateral balance. When the ball is displaced from center, the lateral forces are unbalanced and the airplane will slip or skid in the direction in which the ball is displaced from center.

FLIGHT RECORDER SYSTEMS

A Flight Recorder system is provided. The system records altitude, indicated airspeed, vertical acceleration, compass heading and elapsed time. The copilot's pitot and static systems supply reference pressures for the operation of the flight recorder. The copilot's RMI supplies the input compass signals.

V.G.H. Recorder

Airplanes, serial numbers N812TW and N813TW, have provisions for installing a V.G.H. (velocity-gravity-height) recorder and acceleration transmitter.

V.G. Recorder

Airplanes, serial numbers N814TW through N817TW, have provisions for the installation of a V.G. (velocity-gravity) recorder.

Flight Recorder Audible Signal

A test switch is provided that will introduce the escapement beat signal into the interphone system when the Flight Recorder is operating properly and the test switch is activated.

ELECTRICAL POWER SOURCES

The flight and navigation instruments are powered from the following general electrical sources: (For detailed information, consult the WIRING DIAGRAM MANUAL.)

115-volt ac, 400 cps

Pilot's and copilot's flight director system, Bendix (emergency)
Pilot's integrated flight instrument system, KIFIS
No. 1 VHF NAV RCVR, No. 1 GLIDE SLOPE RCVR, No. 1 ADF RCVR MARKER BEACON, WX RADAR, No. 1 ATC
Pilot's remote compass
Pilot's turn and slip indicator
Pilot's instrument-bus fail relay
Flight Recorder
Pilot's and copilot's flight director system, Bendix
Copilot's integrated flight and instrument system, KIFIS
No. 2 VHF NAV RCVR, No. 2 GLIDE SLOPE RCVR, No. 2 ADF RCVR, No. 2 ATC
Copilot's remote compass
Copilot's turn and slip indicator
Copilot's instrument-bus fail relay
Main instrument transformer

28-volt dc power

No. 1 normal radio dc bus, MARKER BEACON, No. 1 ATC, WX RADAR, TERRAIN WARNING
No. 2 normal radio dc bus, No. 2 VHF NAV, No. 2 ADF, No. 2 ATC
Ram air temperature indicator
26-volt ac instrument-bus control and indicator
Essential and emergency dc radio buses
AC instrument power fail lights
Magnetic compass light
Emergency bus-off relay
No. 1 VHF NAV RCVR
No. 1 ADF RCVR
V.G.H. Recorder (when installed)
V.G. Recorder (when installed)

Section 19

RADIO AND COMMUNICATION SYSTEMS

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RADIO AND COMMUNICATION SYSTEMS

GENERAL RADIO AND COMMUNICATION SYSTEMS

The communication systems include two VHF radio communication systems, a two-channel selective-call system, a high-fidelity public address system, a flight and service interphone system, and provisions for installing a dual HF communication system, a tape reproducer, and a passenger-call light system.

Receivers, transmitters and audio amplifiers are installed on shock-mounted shelves in the electronics compartment located below the flight deck floor. This compartment is accessible during flight emergencies through a door in the floor of the forward coat closet. The selective-call unit is mounted behind the pilot on the aft bulkhead of the flight compartment to provide accessibility for setting the call-code channels.

Controls for the radio communication systems are located on the pilot's and co-pilot's consoles, on the forward and aft areas of the pilots' pedestal, on the pilot's overhead switch panel and at the flight engineer's station. Interphone and public address system controls are located in the flight compartment and in the passenger cabin. Jacks and jackboxes for service interphone communication are provided in the flight compartment, electronics compartment, wheel wells, and in the engine pods and other service areas of the airplane.

RADIO POWER CONTROL PANEL

Two ac and two dc airplane electrical buses supply power to the radio communication and navigation systems. Electric power from these buses is controlled by two ON-OFF toggle switches in the RADIO POWER panel located on the pilots' overhead switch panel.

The NORMAL switch connects the No. 3 essential ac and the 28-volt dc essential buses of the electrical system of the airplane to the No. 1 and No. 2 normal ac and dc radio buses. The HF-1 communication system is powered by the emergency radio bus; the HF-2 system is powered by the No. 2 normal ac and dc radio buses. The VHF-2 communication system is connected to the No. 2 normal dc radio bus. The selective-call system receives ac and dc power from the No. 2 normal ac and dc radio buses.

The ESSENTIAL switch connects the pilot's ac essential and the dc emergency buses of the electrical system to the essential ac and dc radio buses. The VHF-1 receiver and transmitter are connected to the essential dc radio bus. The ESSENTIAL switch can be overridden by two circuit breakers in the main circuit breaker panel in the flight compartment.

AUDIO SELECTOR PANELS

An audio selector panel is provided for the pilot, for the copilot, for the flight engineer and an additional panel is installed in the electronics compartment for maintenance use and in-flight emergencies. The pilot's and copilot's selector panels are located on their respective consoles and the flight engineer's selector panel is on the lower portion of his panel. All four panels are identical.

Each audio selector panel contains toggle switches for selecting the audio output of the various radio navigation and communication systems as well as the interphone. A transistorized isolation amplifier enables the operator to monitor the audio output of any number of the available audio systems. Selection at one panel does not affect selection at another panel. A saftied EMERGENCY - NORMAL toggle switch on the panel can be used to monitor the output of any selected receiver if the isolation amplifier fails. In the EMERGENCY position, only one audio system can be monitored at a time.

A SPEAKER toggle switch allows pilot and copilot to monitor the audio circuits without wearing headsets by switching on the overhead speakers. A 3-position FILTER rotary selector switch enables the operator to separate the voice from the radio range signals when monitoring the audio output of the ADF receivers. A MIC SEL rotary switch, located in the lower center of the panel, is used to connect the microphone output to the VHF-1, VHF-2, interphone, HF-1, or HF-2 transmitters. An audio VOL knob on the microphone selector switch is used to adjust the volume of the audio output. A MICROPHONE SELECTOR rotary switch enables the operator to select the microphone in the smoke mask, the oxygen mask, or the boom microphone. The pilot, copilot and flight engineer hand microphones are connected directly to the audio selector panel and are not affected by operation of the microphone selector switch.

RADIO COMMUNICATION SYSTEMS

The airplane is equipped with two VHF radio communication systems and full provisions are made for the installation of two HF systems. The VHF systems provide static-free reception with range limited to line-of-sight.

VHF Communication Systems

Two separate VHF communication systems provide two-way, static-free communication with FAA and company radio stations. Each system includes a crystal-controlled transmitter and a receiver combination designed to operate in 360 channels between 118.0 and 135.9 mc.

The VHF systems are designated VHF-1 and VHF-2. Receiver audio output is selected by the two VHF toggle switches on the audio selector panels. The microphones can be connected to either system by the MIC SEL switches.

Each VHF system includes a receiver, transmitter, remote control panel, power relay, and an antenna.

The VHF-1 system is connected to a blade antenna installed on the underside of the forward fuselage; the VHF-2 system is connected to a flush antenna in the dorsal fairing on the top of the fuselage. The remote control panel is installed on the pilots' pedestal.

The VHF-1 receiver and transmitter are designed to operate in either a single-channel or double-channel mode. In double-channel operation, the transmitter operates on a frequency six mc above that of the receiver. During double-channel operation, the receiver is tuned to the selected frequency and the transmitter adjusts automatically to a frequency six mc higher. Double-channel operation is limited to receiver frequencies from 118.0 to 120.95 mc, and from 127.0 to 129.95 mc.

VHF Control Panels

A control panel is provided for each VHF communication system. The controls for the VHF-1 communication and the VOR-1 navigation systems are combined in a single panel. An identical panel contains the controls for the VHF/VOR-2 communication and navigation system. A white line through the center of the panels separates the two functions. Both panels are installed on the pilots' pedestal.

The controls for each VHF communication system are located on the upper half of the control panels. The control knobs are white and the frequency window is marked VHF. The selector knob at the left rotates an 18-position switch for selecting the whole numbers in the frequency range from 118 to 135 mc. The knob at the right controls a 10-position switch for selecting the decimal settings from 0.0 to 0.9. This arrangement provides for tuning each radio set to 180 different frequencies spaced 100 kc apart in the band between 118.0 and 135.9 mc.

A toggle switch labeled DC-SC is located adjacent to the left selector knob. This switch is used to select the double-channel or single-channel mode of operation. A small SENS knob mounted on the top of the left selector knob governs the sensitivity of the receiver.

Push-to-Talk Microphone Switches

Thumb-operated push-to-talk switches on the aft outboard horn of each control wheel are provided for radio or interphone communication. A push-to-talk switch is located on the flight engineer's microphone selector panel.

HF Communication Systems

Provisions have been made for installing two HF communication systems to provide communication in the frequency range between 2 and 18.5 mc. HF communication can be used advantageously beyond the normal VHF range or when operating in countries with widely scattered communication stations.

Switches for the HF systems are installed on the audio panel and space for the HF control panels is provided on the pilots' pedestal. The tip of the vertical stabilizer constitutes the HF antenna for both HF systems and is used for both transmitting and receiving. The HF antenna coupler, normally installed in the vertical stabilizer, automatically tunes the antenna to the selected transmitting frequency.

SELECTIVE CALLING SYSTEM

The SELCAL control panel is located on the pilots' pedestal and contains two receiver selector switches and a flashing light associated with each switch. The switches, marked VHF-1, VHF-2, HF-1 and HF-2, are used to connect the audio output of any two communications receivers to two detector channels in the SELCAL unit. When the airplane is being called on the frequency of one of the selected radio receivers, the corresponding light will flash and a chime will sound. Before answering the call, the pilot can activate a reset switch to extinguish they light and silence the chime by depressing the light. The system is then ready for the next call.

SELCAL Airborne Unit

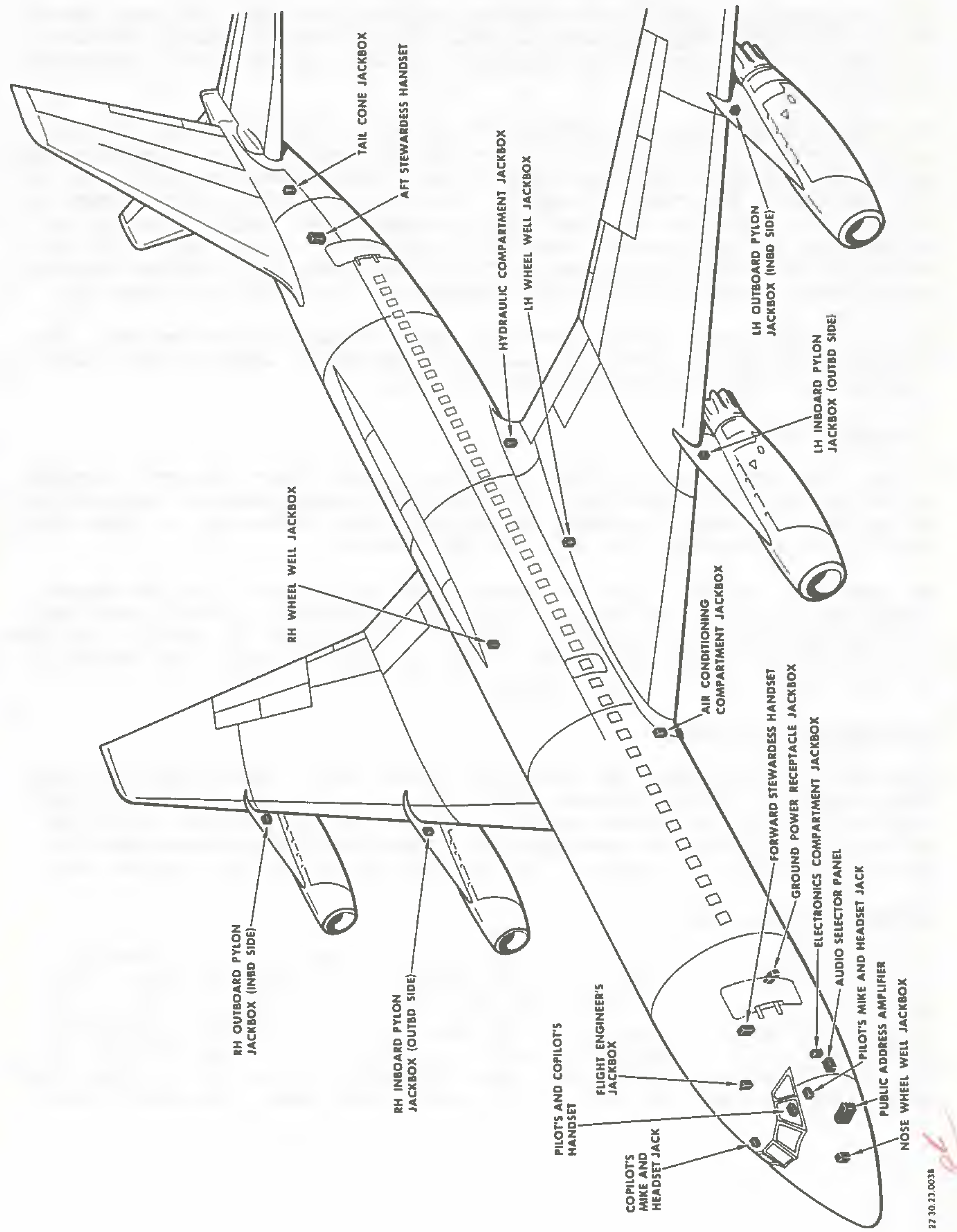
The SELCAL airborne unit consists of two decoder channels enclosed in a metal case mounted on the aft bulkhead in the flight compartment. Removing the cover plate exposes eight selector knobs on the front panel of the case, a set of four knobs for each channel. Each knob rotates through 12 lettered positions from A to M omitting the letter I. The code call for each channel is preset by positioning the selector knobs to correspond to the code assigned to the airplane, starting at the upper left corner of each set of knobs.

The radio receivers receive SELCAL tone signals from ground stations and relay the signals to the decoders of the SELCAL unit. If a coded tone signal is received which corresponds to the code set in one of the decoders of the airborne SELCAL unit, the corresponding light and chime circuits will be energized and the pilots alerted to the incoming call. Each preset decoder channel will respond to only one of 1438 possible tone codes.

The SELCAL chime is mounted in the overhead of the flight compartment adjacent to the interphone chime. Each chime is distinguished from the other by a different distinctive tone.

INTERPHONE SYSTEM

An interphone system is provided for communication between crew members during flight and between members of the ground crew when the airplane is being serviced. Interphone communication is possible from each flight crew station, the fore and aft cabin attendants' stations, the electronics compartment, the nose wheel well, and the principal points in and around the airplane (see Figure 19-1). The crew can converse with one another using the same microphones and



Flight and Service Interphone Locations Diagram
Figure 19-1

audio equipment as used for radio transmitting and receiving. The cabin attendants' stations are equipped with handsets. The service interphone locations are provided with jackboxes for plugging in headsets and microphones or handset combinations.

The interphone system is connected to the emergency dc radio bus and is turned on whenever there is dc available at the airplane emergency bus. Communications are established at the pilot's, copilot's and flight engineer's stations and in the electronics compartment through the switches on the aduio selector panel at each station. The audio is turned on by placing the INT toggle switch in the ON position; the microphone is connected to the system by rotating the MIC SEL switch on the audio selector panel to the INT position. The remaining stations are connected automatically when the communications equipment is plugged in.

The interphone system can be operated in either a flight or service mode. The operating mode is selected by a 2-position ON-OFF toggle switch on the flight engineer's interphone microphone selector panel.

Interphone Flight Mode

In the flight mode, only the pilot's, copilot's, and flight engineer's stations and the outlets in the electronics and nose wheel compartments are connected in the system. The system is placed in the flight mode by placing the flight engineer's service interhpone switch in the OFF position.

The nose wheel outlet is used for communication between the flight and ground crews when the engines are being started. The electronics outlet is used when checking the electronics equipment either in flight or on the ground. This mode of operation is normally used in flight for communication among the members of the crew including the cabin attendants.

Interphone Service Mode

The service mode of operation enables the ground crew to communicate with each other from widely separated areas around and in the airplane. When the flight engineer's service interphone switch is placed in the ON position, all of the outlets in the system are interconnected. Communications are established at the service outlets by plugging in a microphone and headset or a handset unit.

Interphone Call System

A call system is provided to notify the flight crew and the cabin attendants when they are wanted on the interphone system. A pilot call combination light and switch is installed on each cabin attendant's panel. When the switch is depressed, a light will glow and a chime will sound in the cockpit indicating a call from one of the cabin attendants. A STEW call switch is installed in the pilots' overhead switch panel. When this switch is depressed, a light glows on each cabin attendant's panel and the cabin chimes sound, indicating a call

from the flight compartment. These lights remain on until a cabin attendant removes a handset from it's cradle at either control panel.

Handsets

Three handsets are provided; one in the flight compartment mounted on the aft side of the pilots' pedestal and one at each attendant's station in the cabin.

PUBLIC ADDRESS SYSTEM

The public address (PA) system enables the pilots and cabin attendants to address the passengers over the loudspeaker system. Provisions are also made for playing magnetic tape recordings through the public address system amplifier.

The PA system consists of a pilot's handset and control panel installation, forward and aft cabin handset and control panel installations, a priority relay, a high fidelity public address amplifier, and 23 loudspeakers installed in the hatracks in the cabin.

Pilots' voice communications have priority over music and voice announcements from either of the cabin control positions. The forward cabin control position has priority over music and voice announcements from the aft cabin control position. The aft cabin control position has priority over music for making voice announcements.

Public Address Control Panels

A PA control panel is provided for the pilots and each of the two cabin attendants. Each panel carries a vu-meter, volume controls, and call indicator lights. The panels are edge-lighted by lamps in the panels.

The pilots' control panel is installed on the left aft part of the control pedestal. This panel has a volume control switch for microphone audio only. An amber light PUSH PA switch connects the pilot's handset with the public address system. Communications from the panel interrupt cabin music and voice communications being made from either of the cabin PA positions.

The forward cabin PA control panel is installed in the forward buffet area. This panel contains a MUSIC ON-OFF switch and the volume control for adjusting the level of music in the cabin. A voice volume control adjusts the audio level for voice announcements. PILOT CALL and HOSTESS CALL warning lights indicate when the position is being paged by another position. The call light will extinguish automatically when the handset is removed from the cradle.

The aft cabin PA control panel is installed in the aft buffet area. This control panel contains a volume control for handset voice announcements, PILOT CALL and HOSTESS CALL warning lights, and a PUSH PA control switch.

Public Address Handsets

A handset is installed at each control panel. The handset is normally connected to the interphone system. To talk over the PA system, the operator must depress the amber light PUSH PA switch on the PA control panel. The amber light will illuminate indicating that the handset is connected to the PA system. When the handset is restored to the holder, the handset is automatically returned to the interphone system.

PASSENGER CALL SYSTEM

Pushbutton switches and light assemblies installed near the passenger seats and in the lavatories enable the passengers to call the cabin attendants. A call by a passenger illuminates a light near the passenger area, call lights on the forward and aft cabin attendants' panels, and sounds a chime in the cabin calling near each buffet area.

Forward and aft cabin attendant's light panels are identical. Passenger call lights are grouped by rows of seats. Three lights indicate the general area in which the calling passenger is seated. Two lights indicate calls from the forward and aft lavatories. One light signals a call from the flight compartment and one light indicates a call from either cabin attendant's position to the other cabin attendant.

Each light is part of a switch assembly. Pulling the light extinguishes it and resets the system for another call. Cabin light control panels are interconnected so that resetting a light at one panel resets the other panel.

ELECTRICAL POWER SOURCES

The radio and communications equipment receives electrical power from four ac and four dc radio buses. AC power is supplied to the ac radio buses by the No. 3 essential ac bus and the pilot's ac essential bus of the electrical system of the airplane. DC power is provided for the dc radio buses by the 28-volt dc essential and 28-volt dc emergency buses.

Distribution of electric power to the radio buses follows.

NO. 3 ESSENTIAL AC BUS
- No. 1 normal ac radio bus
- No. 2 normal ac radio bus

PILOT'S AC ESSENTIAL BUS
- Essential ac radio bus
- Emergency ac radio bus

28-VOLT DC ESSENTIAL BUS
- No. 1 normal dc radio bus
- No. 2 normal dc radio bus

28-VOLT DC EMERGENCY BUS
    Essential dc radio bus
    Emergency dc radio bus

Electric power distribution from the radio buses to the radio and communication systems can be determined by consulting the WIRING DIAGRAM MANUAL.

Section 20

MISCELLANEOUS SYSTEMS

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MISCELLANEOUS SYSTEMS

AIRPLANE LIGHTING SYSTEM

The airplane lighting system includes all the necessary lights for flight crew members, passengers and maintenance personnel, and airplane navigation requirements. The lighting systems are divided into four subsystems:

1. Flight compartment lights.
2. Passenger and lounge compartment lights.
3. Cargo and miscellaneous compartment lights.
4. Exterior airplane lights.

Alternating and direct current power sources are used for the various subsystems. Circuit breakers are provided to protect each indivudial light circuit from any electrical overload condition that may occur.

FLIGHT COMPARTMENT LIGHTS

Two dome lights are located in the flight compartment ceiling aft of the pilots' overhead switch panel. Each dome light has a red and white bulb. A three position COMPT LT, RED-OFF-WHITE, toggle switch on the overhead switch panel controls the lights. A pushbutton thunderstorm switch is mounted in each pilot's control wheel. When pressed, these switches will illuminate the white dome lights regardless of the position of the toggle switch. When the thunderstorm switches are released, the dome lights will return to the illumination mode called for by the toggle switch position.

Instrument Panel Lights

The pilots' instrument panel is illuminated by sixteen red and twelve white floodlights mounted in the glare shield above the instrument panel. Three high intensity white fluorescent floodlights, located in the glare shield provide additional instrument panel illumination. Eleven of the sixteen red lights illuminate the pilot's flight instrument panel and the engine instrument panel. The eleven red lights are controlled by a rheostat type PILOT PANEL RED LIGHT switch located on the pilot's console. The remaining five red lights illuminate the copilot's flight instrument panel and are controlled by a rheostat type CO-PILOT PANEL RED LIGHT switch located on the copilot's flight instrument panel. The twelve white floodlights in the glare shield are controlled by a rheostat type PANEL WHITE LIGHT switch located on the pilot's flight instrument panel. This switch also illuminates the high intensity white fluorescent floodlights when rotated to the full BRT position.

Pilot's Console Lights

The pilot's and copilot's consoles are each internally illuminated by four red lights in the console and controlled by a rheostat type CONSOLE LTS switch on

each console. Four red floodlights are provided above each console. These lights are controlled by a three position BRT-OFF-DIM toggle switch mounted on each console panel. Two additional lights, for the oxygen regulators on each console, are connected with the integral red lights circuits.

Pilots' Overhead Switch Panel Lights

Each section of the pilots' overhead switch panel is internally illuminated by red lights mounted in the panel. A rheostat type OVERHEAD PANEL LIGHTS switch on the overhead switch panel controls the light intensity.

Pilots' Pedestal Lights

Each section of the pilots' pedestal is internally illuminated by red lights mounted in the panel. The lights are controlled by a rheostat type PEDESTAL LIGHTS switch located on the pilots' pedestal. A red semispotlight is located in the flight compartment ceiling at station 300 to provide additional pedestal and overhead switch panel illumination. A three position BRIGHT-OFF-DIM toggle switch, PEDESTAL FLOOD, on the overhead switch panel controls the floodlight.

Flight Engineer's Panel Lights

Each section of the flight engineeer's instrument panel is internally illuminated by red lights mounted in the panel. A rheostat type FLIGHT ENGINEERS PANEL LIGHTS switch located on the flight engineer's panel controls the light intensity. In addition, the flight engineer's panel has one red and one white floodlight mounted in the flight compartment ceiling at fuselage station 266. These lights are controlled by individual rheostat type ENGINEERS FLOODLIGHT switches located on the aft side of the flight engineer's panel. Lights are also provided to illuminate the circuit breaker panels and the limiter and ac panels under the flight engineer's table.

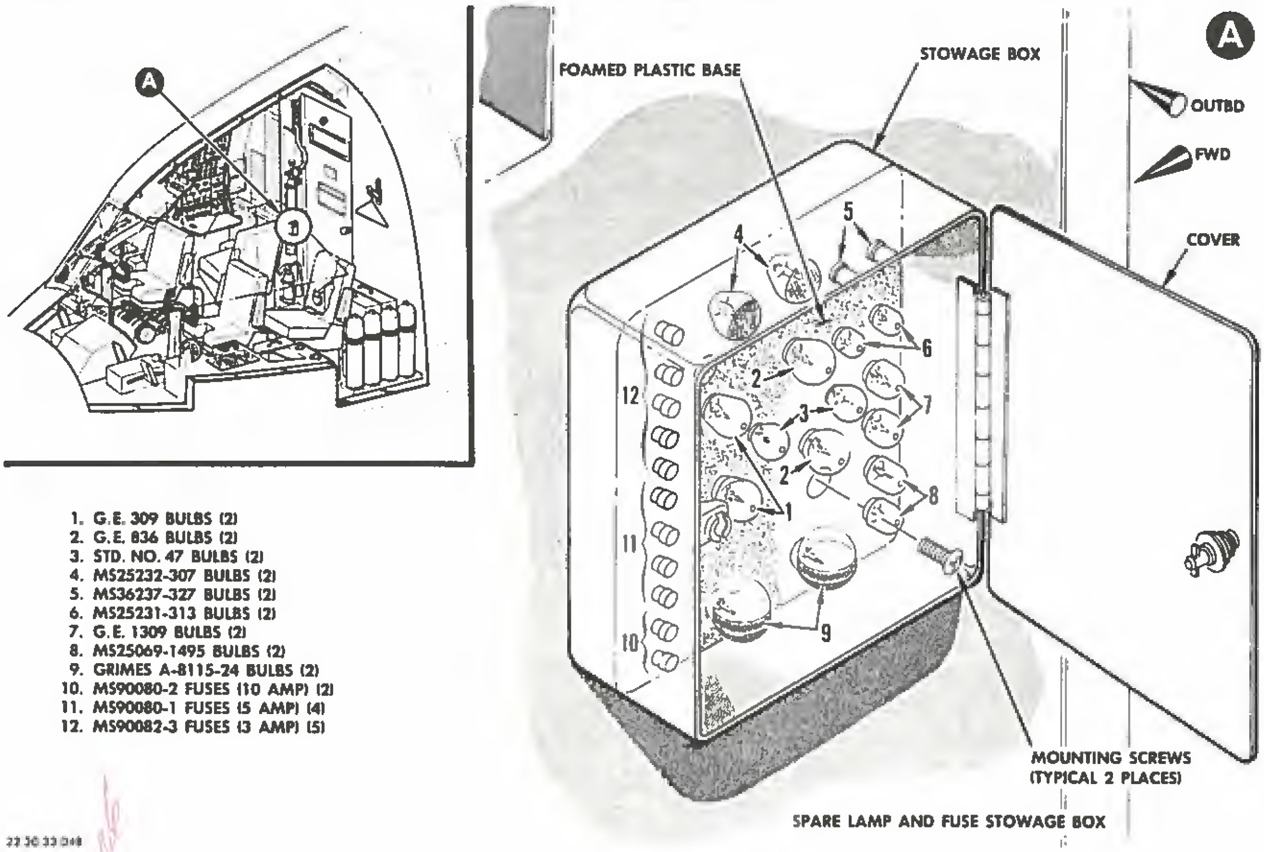
A utility light is also provided on the flight engineer's panel. An integral ON-OFF switch is included on the light assembly.

Map Reading Lights

Map reading lights are provided for the pilot and copilot. The pilot's map reading light is located to the left of the overhead switch panel and the copilot's light is to the right of the switch panel. Each light has a built-in ON-OFF toggle switch and light intensity control knob. Each light can be adjusted from a 14-inch diameter spot to a 2-inch diameter spot when directed on a map in the crew members lap.

Standby Compass Light

A red light is provided for the magnetic standby compass on the glare shield. The rheostat type STANDBY COMPASS LIGHT switch is located on the copilot's flight instrument panel. A spare lamp and fuse stowage box is mounted on the flight compartment aft bulkhead. Spare lamps, two of each type used for the flight compartment lights, are stored in a shock-absorbing, foamed plastic retainer inside the box (see Figure 20-1).



Spare Lamp Stowage Box
Figure 20-1

PASSENGER AND LOUNGE COMPARTMENT LIGHTING

The passenger and lounge compartment lighting includes the cabin general lights, lounge overhead and table lights, aisle lights, passenger reading lights, passenger entrance lights, coat closet lights, emergency lighting system, and lavatory and buffet lights.

Cabin General Lights

Fluorescent lights located in the cabin overhead and above the cabin windows provide general illumination in the passenger cabin. The overhead lighting is indirect and consists of twenty-one, 13-watt flourescent lights mounted on each side of the ceiling. The window lighting consists of twenty-two lights installed on each side of the passenger cabin, one above each pair of windows. All of the windows lights, except the light directly aft of each emergency exit, are 13-watt units. The two flourescent lights directly aft of the left and right emergency exits are 8-watt units.

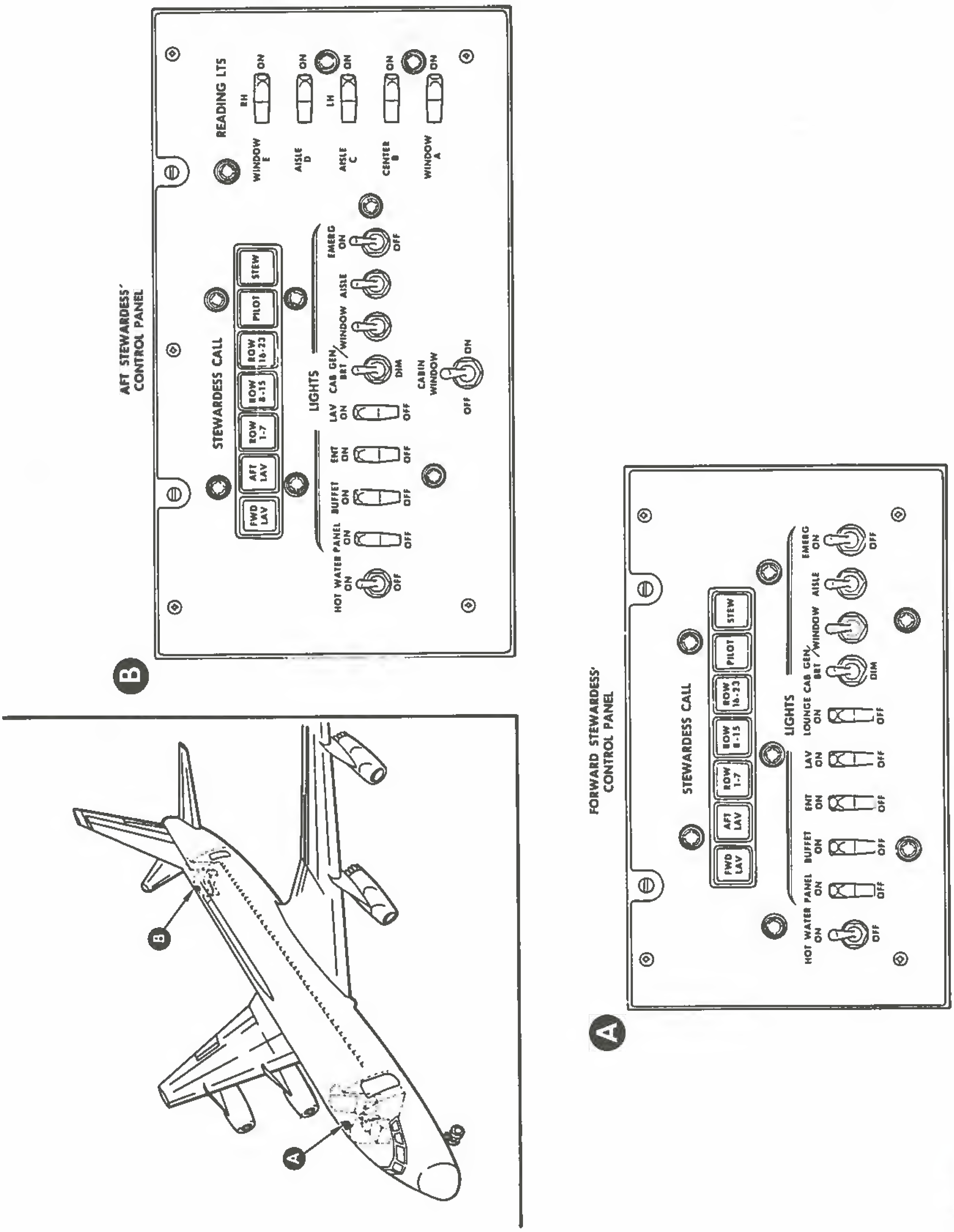
The general cabin light intensity is controlled by an electric motor-driven, dimming rheostat. The driving motor is controlled from either of the two stewardess's panels by a three-position DIM-NEUTRAL-BRIGHT toggle switch, spring-loaded to the NEUTRAL position. The toggle switch is held in the DIM or BRIGHT position until the light intensity is as desired. When the switch is in the DIM position, the final few degrees of rheostat rotation actuate a control switch which turns the cabin lights off. The general cabin lights can be extinguished from the flight compartment by means of an override switch located on the pilots' overhead switch panel. The cabin window lights can be separately controlled, if desired, by an ON-OFF toggle switch on each stewardess's control panel. These switches are wired in a three-way circuit to provide full operation from either panel (see Figure 20-2).

Lounge Overhead Lights and Table Light

Lounge lighting is provided by eight overhead lights and one lounge table lamp. The lounge lights are controlled by a circuit breaker type switch located on the forward stewardess's control panel. The lounge table lamp is electrically connected to the overhead lounge lights and also includes an ON-OFF switch for local control.

Aisle Lights

Six aisle lights are installed in the cabin ceiling to illuminate the aisles between the passenger cabin seats. The aisle lights are controlled by two ON-OFF toggle switches, one on each stewardess's control panel. These switches are wired in a three-way circuit to provide full operation from either panel.



Stewardess's Control Panels
Figure 20-2

Passenger Reading Lights

The passenger reading lights are located in the hat racks, one above each passenger seat. Pushbutton switches (push to illuminate, push again to extinguish) are located adjacent to each light. An override switch, PASSENGER READING LIGHTS in the pilots' overhead switch panel permits the pilots to extinguish the passenger reading lights. Five READING LIGHTS switches on the aft stewardess's panel must be ON to arm the individual control switches.

Passenger Entrance Lights

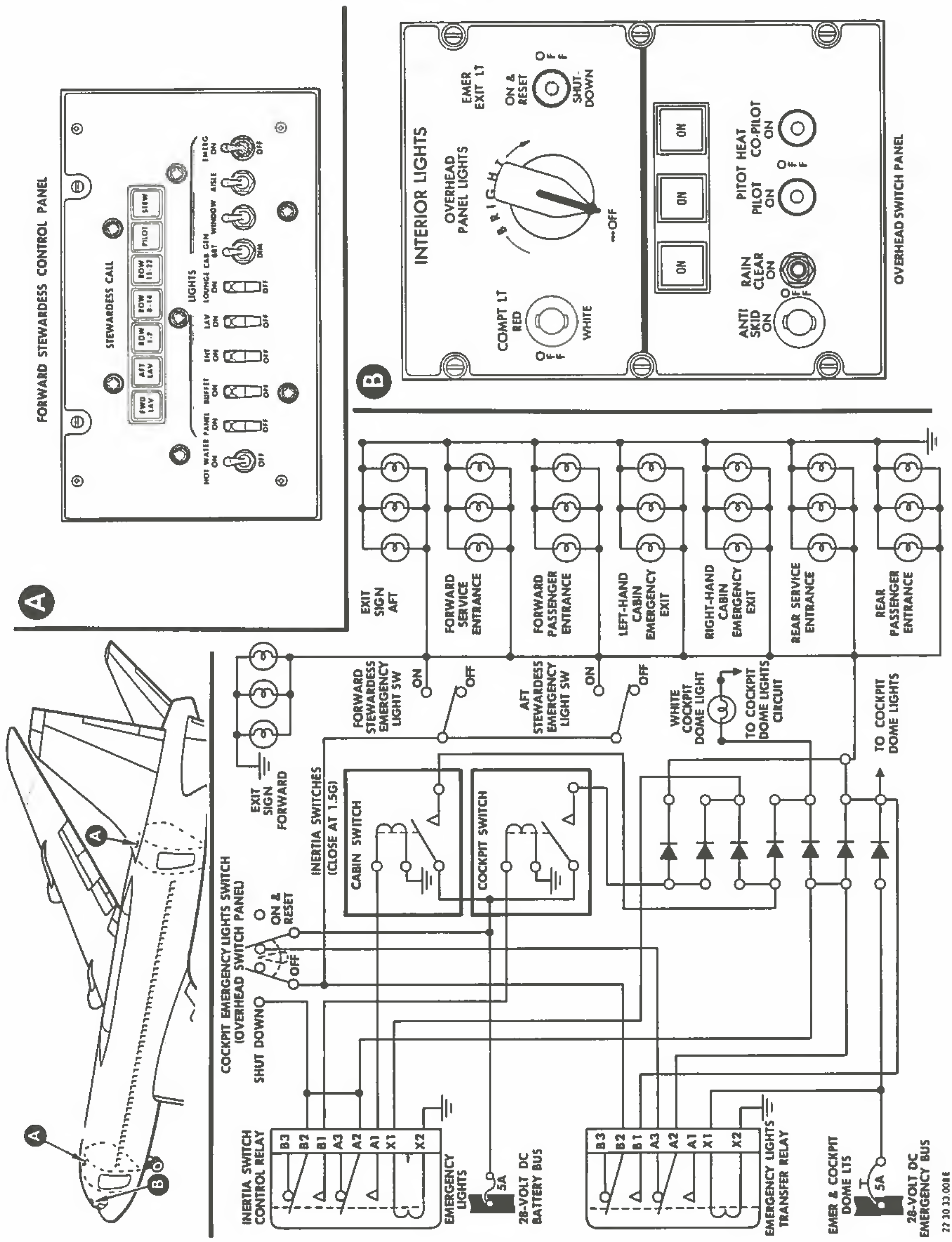
Two threshold lights and two overhead lights are located at each main passenger entrance. The lights are controlled by a switch in the corresponding main entrance door sill and are extinguished when the doors are closed and locked. This switch also controls the DOOR OPEN ABOVE FLOOR warning light in the flight compartment. The lights illuminate when the main entrance doors are opened. An override ENTRANCE, ON-OFF, switch is provided on the respective stewardess's control panel. Also located at each main entrance is a RAMP RECEPTACLE which is supplied with airplane electrical power.

Coat Closet Lights

The forward and aft coat closets are each provided with the 8-watt fluorescent light controlled by a circuit breaker type ON-OFF switch located on the side of each coat closet.

Emergency Lighting System

The emergency lighting system includes the lights at the forward and aft main entrance and service areas, and the left and right passenger cabin emergency exits, and the white dome lights in the flight compartment. These lights may be energized by either the 28-volt dc emergency bus or the battery bus. The EMERGENCY EXIT LIGHT switch in the pilots' overhead switch panel is a three-position, ON and RESET-OFF-SHUTDOWN toggle switch. The forward and aft stewardess's EMERGENCY LIGHTS switches are two-position, ON-OFF toggle switches. When either of the stewardess's emergency switches is placed in the ON position, all emergency lights except the flight compartment dome lights will illuminate. When the flight compartment switch is placed in the ON and RESET position, all emergency lights will illuminate. When the flight compartment switch is placed in the SHUTDOWN position, electrical power from the battery bus is interrupted. In the event of an electrical power failure on the dc emergency bus, a transfer relay deenergizes and transfers battery bus electrical power to the emergency lights. The lights may be extinguished by placing the flight compartment switch in the SHUTDOWN position regardless of the stewardess's panels emergency lighting switch positions (see Figure 20-3).



Emergency Lighting System Schematic
Figure 20-3

Emergency Light Inertia Switches

If a shock force of 1.5 G's or more is experienced, two inertia switches will close and energize the emergency lighting circuit, illuminating all emergency lights from the battery bus.

Lavatory Lights

Three 13-watt flourescent lights in the forward lavatory are controlled by a switch on the forward stewardess's control panel. A lavatory door switch automatically extinguishes two of the three lights when the door is opened. Both aft lavatories have two 12-watt and two 8-watt fluorescent lights each, all controlled by a switch on the aft stewardess's control panel. Each aft lavatory door switch automatically extinguishes three of the lights when the respective door is opened.

Buffet Area Lights

Four lights are located in the ceiling of the forward buffet and two lights in the ceiling of the aft buffet. Switches of the stewardess's panel in their respective areas control the lights.

Cabin Attendant's Control Panel Lights

The forward stewardess's control panel is illuminated by nine integral red lights. A control switch, PANEL, ON-OFF, is located on each panel.

CARGO AND MISCELLANEOUS COMPARTMENT LIGHTING

The cargo and miscellaneous compartment lighting includes the lights in the forward and aft cargo compartments, the electrical and electronic compartments, the air conditioning compartment, main and nose gear wheel wells and the hydraulic and pneumatic compartment. Also included are the lighting circuits for the aft fuselage interior and the pylon refueling panels.

Forward and Aft Cargo Compartment Lights

Three dome lights and one threshold light are located in each forward and aft cargo compartment. The lights are controlled by automatic door switches, which also actuates the DOOR OPEN BELOW FLOOR warning light in the flight compartment.

Electrical and Electronic Compartment Lights

Four lights provide illumination in the electrical and electronic compartment. The lights are controlled by the access door switch or an ON-OFF toggle switch adjacent to the inner surface of the access door. The toggle switch is used when gaining entry to the compartment through the flight compartment floor door. This door is connected into the DOOR OPEN BELOW FLOOR warning light.

Air Conditioning Compartment Lights

Two lights are located in each of the forward and aft air conditioning compartments. The lights are controlled by automatic door switches which also actuate the DOOR OPEN BELOW FLOOR warning light.

Wheel Well Lights

One light is located in each wheel well. These lights are connected to the position light circuit and are armed through the landing gear door "up" and "locked" warning switches. Thus the airplane must be on the ground with position lights ON to illuminate the wheel well lights.

Hydraulic and Pneumatic Compartment Light

One light, automatically controlled by the access door, illuminates the hydraulic and pneumatic compartment. This door is connected into the DOOR OPEN BELOW FLOOR warning light.

Aft Fuselage Lights

Two lights provide illumination of the area aft of the aft pressure bulkhead. The forward light is controlled by an automatic door switch on the forward access door and the aft light is controlled by a similar automatic switch on the aft tail cone access door.

Pylon Refuel Panel Lights

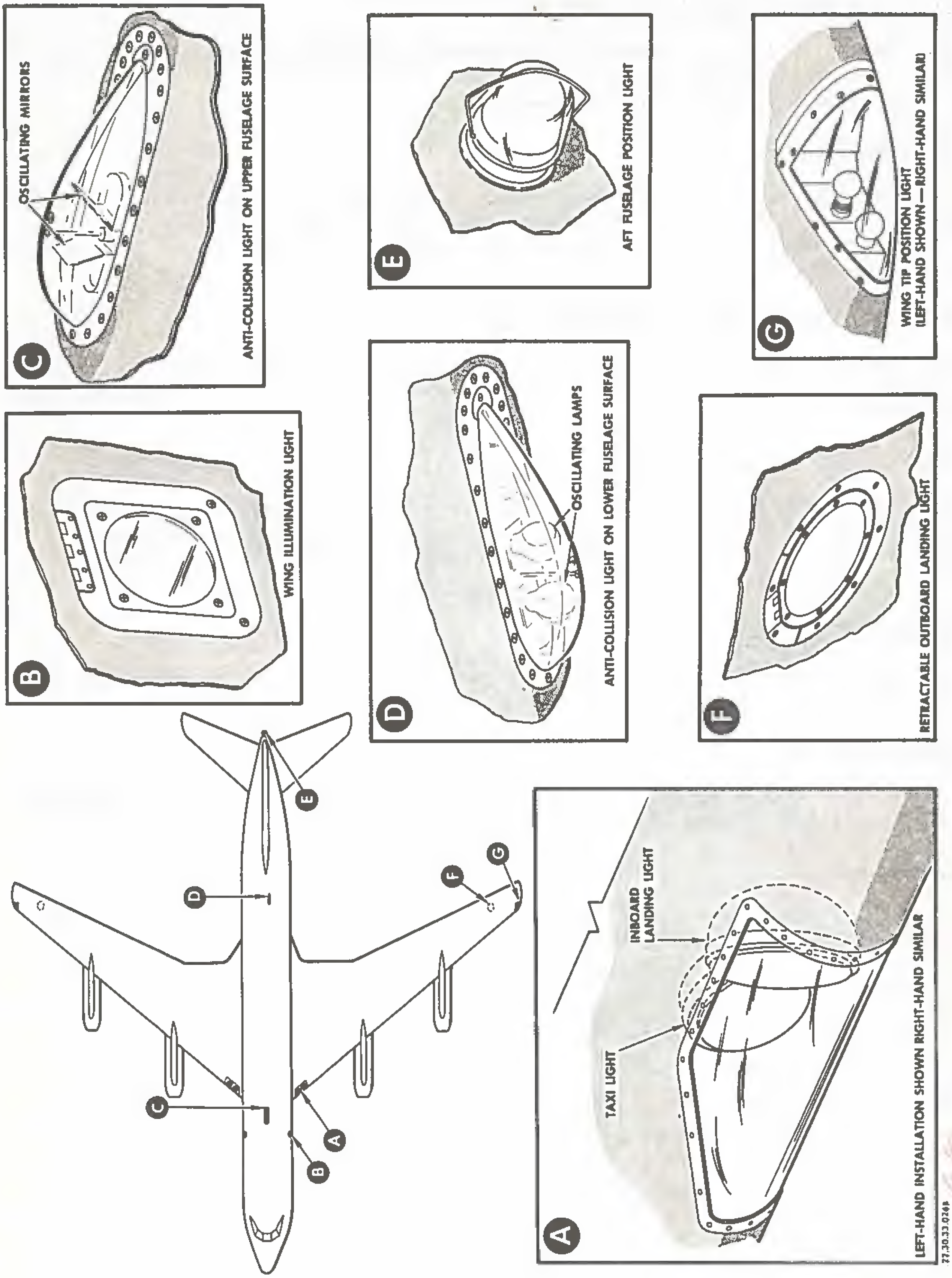
One light illuminates each pylon refuel panel. The lights are controlled by door actuated microswitches.

EXTERIOR LIGHTS

The exterior lights include the landing lights, taxi lights, wing illumination lights, position lights, and the anti-collision lights (see Figure 20-4).

Landing Lights

Two inboard and two outboard landing lights are provided. The fixed position inboard landing lights are located in the leading edge of each wing, just outboard of the fuselage. The outboard landing lights are located in the lower outboard surface of each wing at the rear spar (wing station 697). These lights are retractable units that flush with the wing when not in use. The outboard landing lights use 1000-watt lamps and each light is controlled by a separate three-position EXTEND-OFF-RETRACT toggle switch. In the EXTEND position, the light is extended and illuminated. The light can be held in an extended but extinguished position by placing the switch in the OFF position after extending the light. Two amber warning lights indicate when the outboard landing lights



Exterior Light Assemblies
Figure 20-4

are extended. The inboard landing lights use 600-watt lamps and each light is controlled by a separate ON-OFF toggle switch. All landing light switches and warning lights are located on the pilots' overhead switch panel.

An airspeed pressure switch in the outboard landing light control circuit prevents extension of the landing lights when the airplane speed exceeds ~~150~~ 275 250 knots. The switch signals also cause retraction of the landing lights if the speed exceeds ~~250~~ knots. After such automatic retraction, the landing light control circuit must be reset by momentarily placing either landing light control switch in the RETRACT POSITION. 275 ± 15

Taxi Lights

For taxiing purposes, two 250-watt lamps are located in the leading edge of each wing at station 75.72. Each light is controlled by a separate ON-OFF switch located on the pilots' overhead switch panel.

Wing Illumination Lights

A 100-watt wing illumination light is installed on each side of the fuselage forward of the wing at fuselage station 517. These lights are used to detect ice formation on the wing leading edge. A WING ICE LIGHTS "ON-OFF" switch is located on the pilots' overhead switch panel.

Position Lights

The position lights consist of a green light on the right wing tip, a red light on the left wing tip and a white light on the aft tip of the fuselage. Each wing position light incorporates two light bulbs. An ON-OFF toggle switch located on the pilots' overhead switch panel illuminates the position lights. The position light control switch also controls the dimming of the master warning system lights. A flasher unit is not included in the position light circuitry.

Anticollision Lights

Two 40-watt anticollision lights are provided, one on the upper surface of the fuselage at station 555 and the other on the lower surface of the fuselage at station 1070. These units are separately controlled and operated. Each anticollision light consists of two rotating lights operated by an electric motor. The lights are controlled by switches on the pilots' overhead switch panel.

AIRPLANE LIGHTING SYSTEMS POWER SOURCES

For detailed information, consult the WIRING DIAGRAM MANUAL.

115-Volt AC

Pilot's Instrument Panel and Floodlights
Pilot's Console Panel Lights
Pilot's Overhead Switch Panel Lights
Pedestal Panel Lights
Flight Engineer's Panel Lights and Floodlights
Coat Closet Lights
Lavatory Lights
Cabin General Lights
Outboard Landing Lights
Taxi Lights
Inboard Landing Lights
Anticollision Lights
Passenger Reading Lights

28-Volt AC

Map Reading Lights
Lounge Overhead Lights and Table Lamp
Passenger Entrance Lights
Buffet Area Lights
Cabin Attendant Control Panel Lights
Forward and Aft Cargo Compartment Lights
Electrical and Electronic Compartment Lights
Air Conditioning Compartment Lights
Hydraulic and Pneumatic Compartment Lights
Aft Fuselage Lights
Console Floodlights
Wing Illumination Lights

28-Volt DC

Emergency Lighting System (only when power is not available from the 28-volt dc emergency bus)

Flight Compartment Dome Lights
Standby Compass Lights
Aisle Lights
Wheel Well Lights
Position Lights
Pilot's Overhead Switch Panel Floodlight
Pedestal Floodlight
Pylon Refuel Panel Lights
Passenger Loading Ramp Receptacles

WATER SYSTEM

The water system consists of a titanium water supply tank, a ground service panel which is used in connection with maintenance and servicing of the system, and numerous valves and interconnecting tubing to provide the means of delivering water to the proper places.

Pressurization

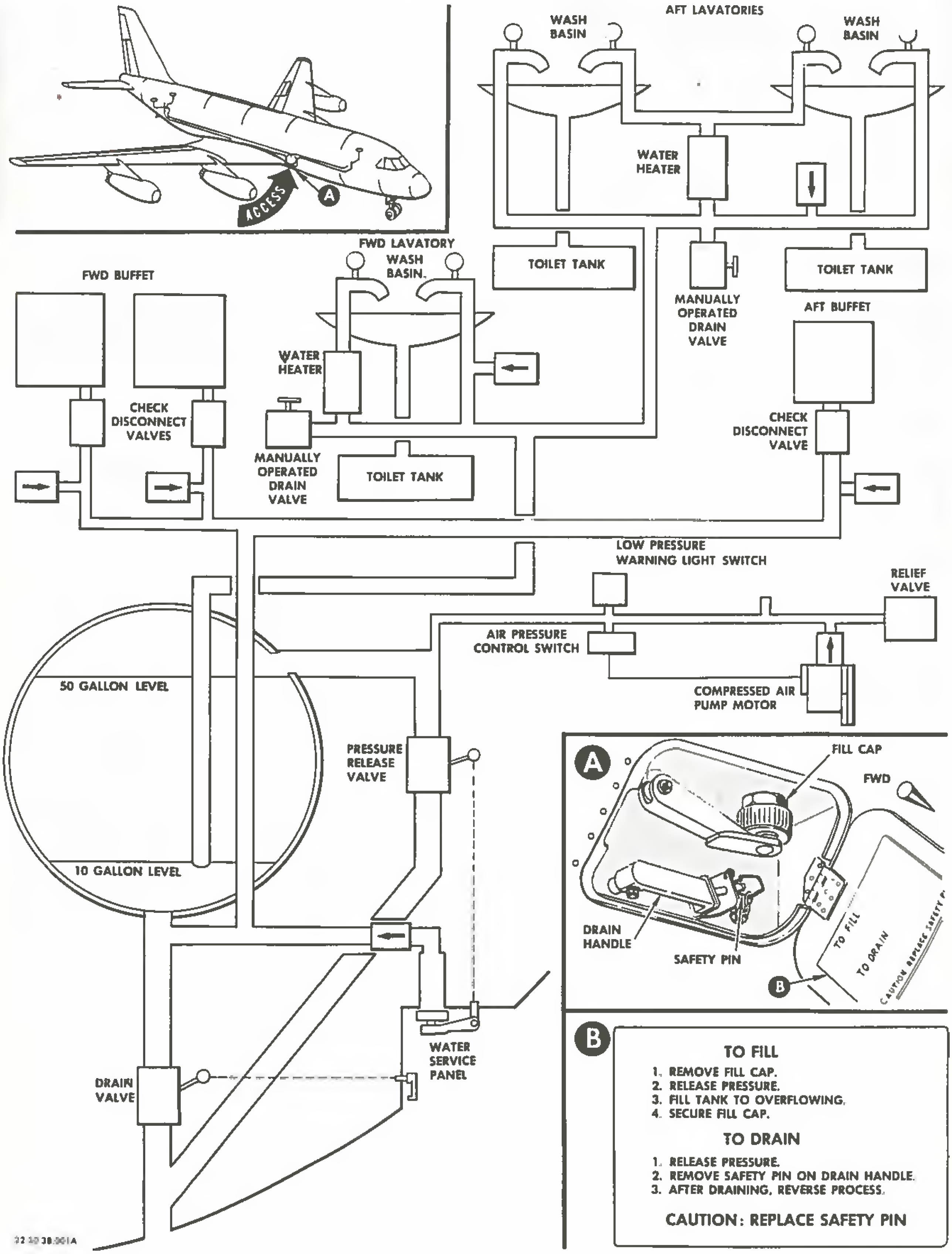
The pressurization circuit consists of a motor-driven air pump, two pump control relays, a manual control switch, a low-pressure warning switch, a low-pressure pump control switch and warning lights (see Figures 20-5 and 20-6).

Heating

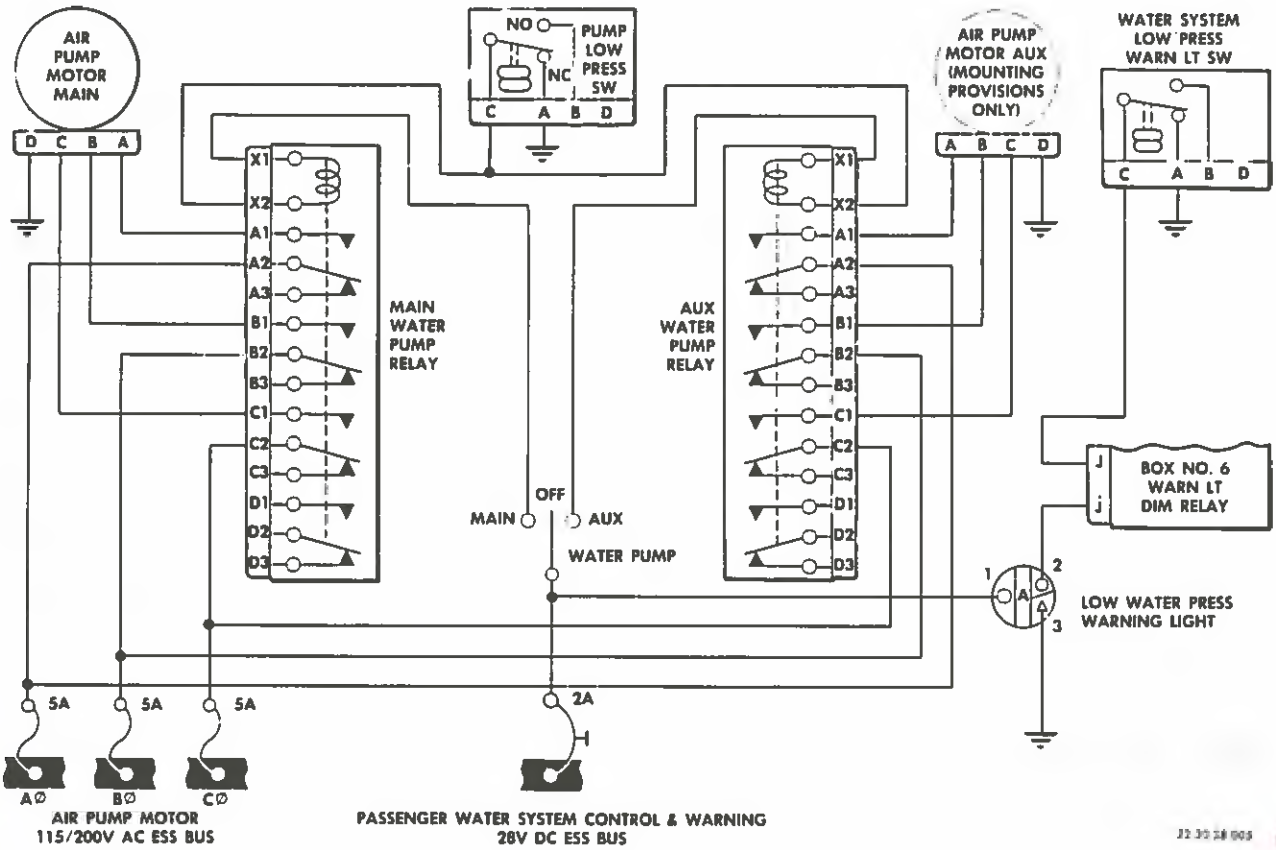
The water heater circuit consists of two water heaters, two control switches, three circuit breakers and interconnecting wiring (see Figure 20-7).

Water System Operation

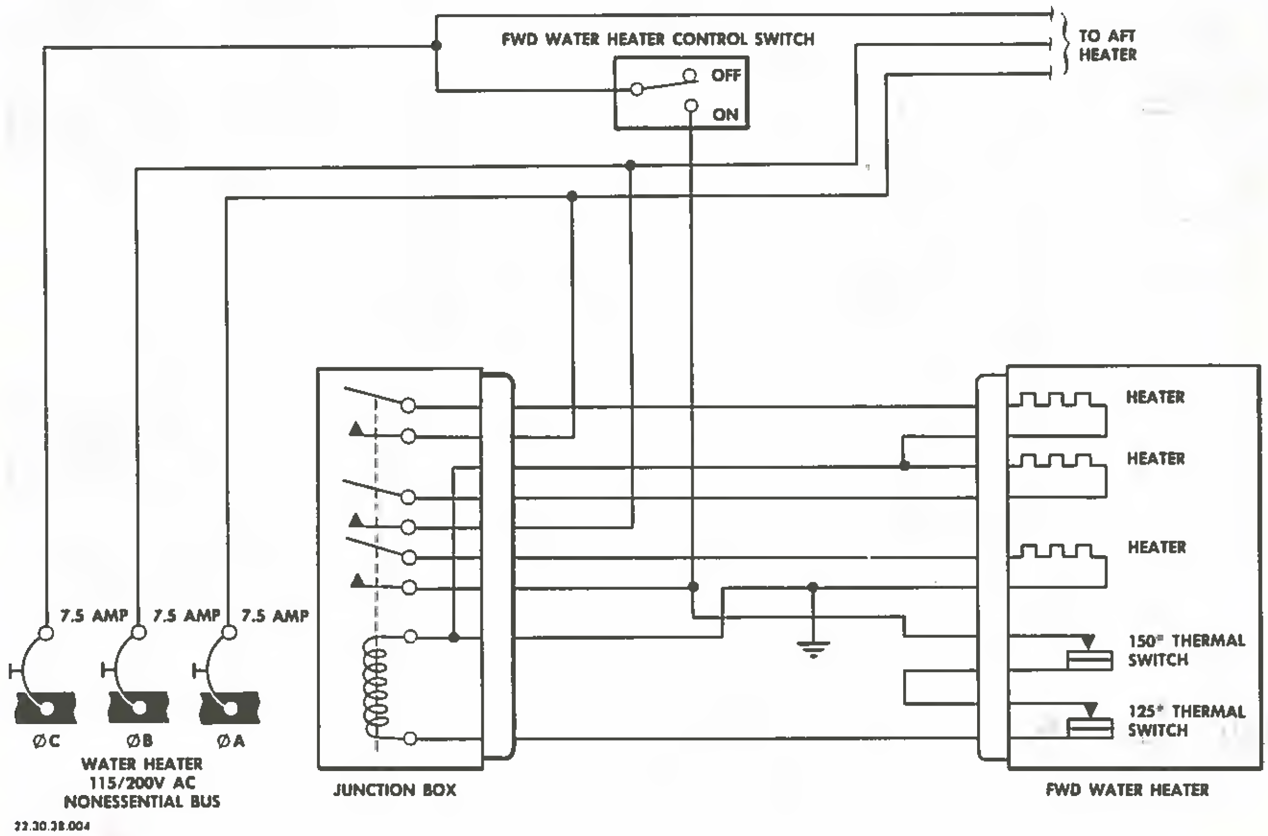
The water for the system is contained under pressure in the supply tank and is routed to the lavatories and buffets on demand when the spigots are opened. Once the pump control switch, located on the flight engineer's panel, is placed in the NORMAL position, the pressurization circuit automatically maintains system pressure through the operation of the low-pressure pump control switch. The low-pressure warning switch monitors the pressurization circuit by illuminating the warning light when the system pressure is too low. As the system water supply is depleted, the air pressure decreases until it reaches a certain point. The decreased pressure allows the low-pressure pump control switch to complete the control circuit for turning on the air pump motor and rebuilding the system pressure.



Water System Schematic
Figure 20-5



Water Pressurization System Circuit
Figure 20-6



Water Heater Circuit
Figure 20-7

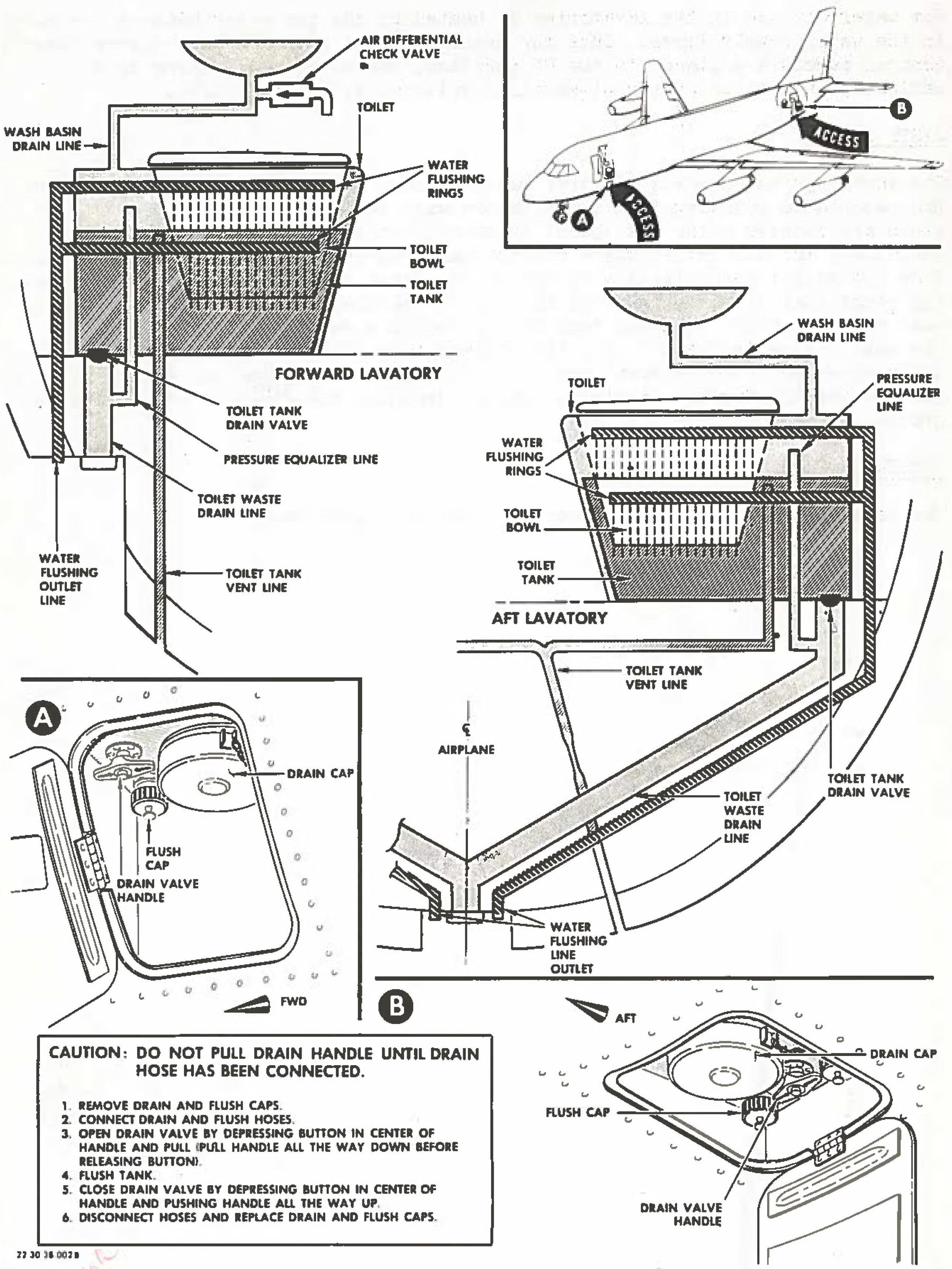
Hot water for use in the lavatories is heated by the two water heaters installed in the water supply lines. Once the heater control switches on the stewardess's control panels are placed in the ON position, the water temperature is automatically maintained by two bi-metallic switches within the units.

WASTE SYSTEM

The waste system consists of three waste storage tanks, two service panels and the associated plumbing from the lavatory wash basins (see Figure 20-8). The tanks are located below the toilet in each lavatory and are used as storage containers for all toilet waste and for waste water from the wash basins. Waste from the toilet goes directly to the storage tank during normal use, and flushing rings that are integral with the top of the bowl and tank are utilized to wash waste from the bowl and tank during ground servicing. Waste water from the wash basins is drained into the storage tank when the sink stopper is manually operated. Two external ground service panels, one for the aft lavatories and one for the forward lavatory, provide draining and flushing facilities for ground servicing.

PASSENGER WARNING LIGHT SYSTEM

The passenger warning light system is shown in Figure 20-9.



CAUTION: DO NOT PULL DRAIN HANDLE UNTIL DRAIN HOSE HAS BEEN CONNECTED.

1. REMOVE DRAIN AND FLUSH CAPS.
2. CONNECT DRAIN AND FLUSH HOSES.
3. OPEN DRAIN VALVE BY DEPRESSING BUTTON IN CENTER OF HANDLE AND PULL (PULL HANDLE ALL THE WAY DOWN BEFORE RELEASING BUTTON).
4. FLUSH TANK.
5. CLOSE DRAIN VALVE BY DEPRESSING BUTTON IN CENTER OF HANDLE AND PUSHING HANDLE ALL THE WAY UP.
6. DISCONNECT HOSES AND REPLACE DRAIN AND FLUSH CAPS.

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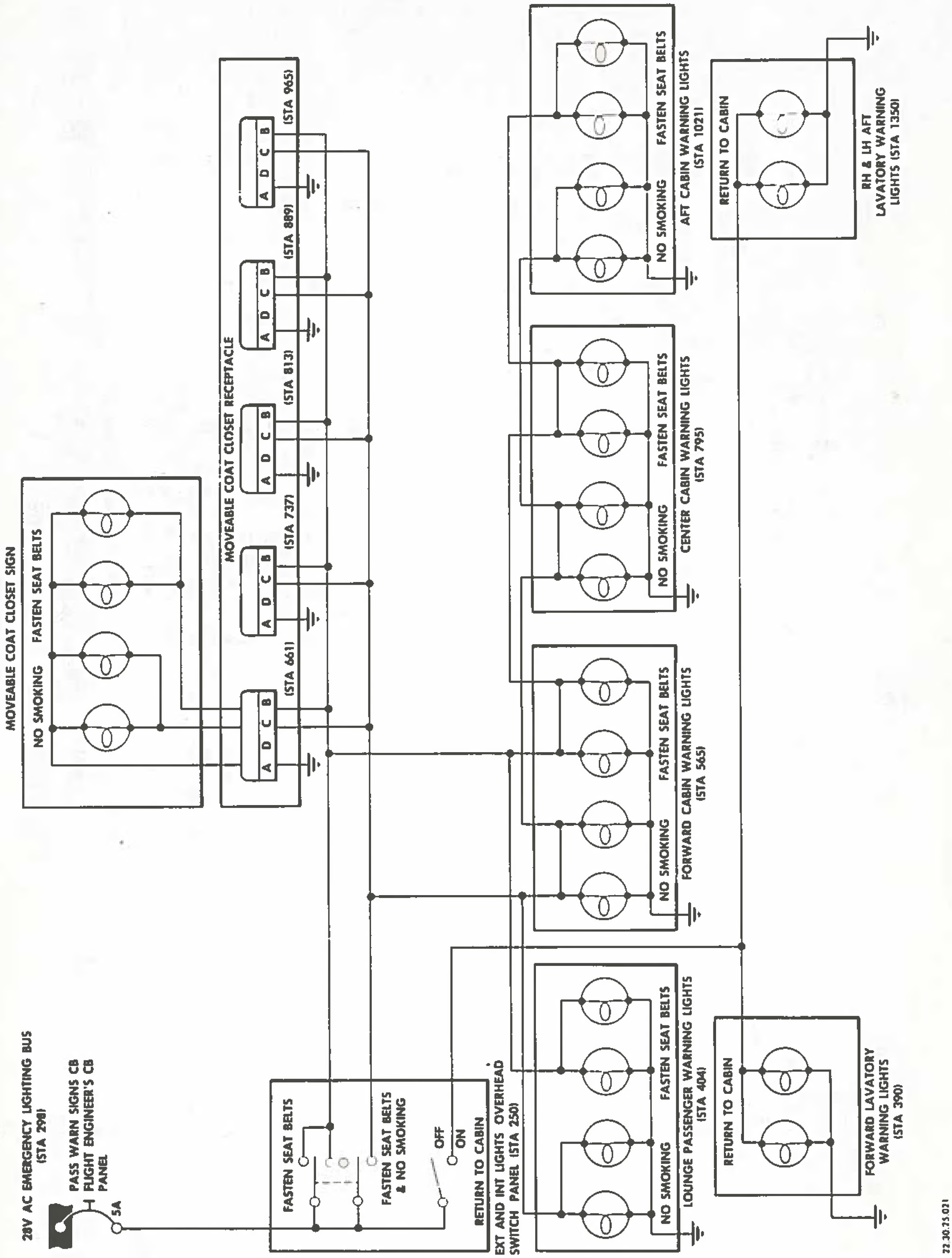
Waste System Schematic
Figure 20-8



Passenger Warning Lights Circuit
Figure 20-9